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GROUND INSTALLATIONS FOR DEVELOPMENT, QUALIFICATION, AND CHECKOUT TESTING OF THE APOLLO SPACE VEHICLE

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GROUND INSTALLATIONS FOR DEVELOPMENT, QUALIFICATION, AND CHECKOUT TESTING OF THE APOLLO SPACE VEHICLE

The objective of the Apollo Program is to land men on the moon and return them safely to earth in this decade, and through accomplishment of this objective, to develop a broad capability for manned space flight. The program was initiated by the late President John F. Kennedy in 1961. Today the design and facility buildup phases have substantially been completed and the program is well into the ground test phase, as indicated in figure 1. The development and operation of the Apollo system is carried out by three major laboratories of the National Aeronautics and Space Administration (NASA), and by approximately 30 major prime contractors and their myriad of subcontractors under overall program management of the Apollo Program Office in Washington, D.C.

This paper will outline the ground test plan for the Apollo space vehicle and will describe the requirements and capabilities of some of the major ground installations utilized in developing, qualifying, and performing factory checkout and acceptance testing of the Apollo space vehicle. These ground system capabilities are significant to Apollo follow-on programs as well, inasmuch as they represent an existing base from which post-Apollo ground development activity can proceed.

The space vehicle for accomplishing the Apollo lunar mission is designated the Apollo-Saturn V space vehicle and is illustrated in figure 2. Total launch weight is about 6,000,000 pounds and first stage thrust is 7,500,000 pounds. The Saturn V launch vehicle consists of three stages: the first stage, designated S-IC, has five LOX-kerosene F-1 engines of 1,500,000 pounds thrust each; the second stage, designated S-II, has five hydrogen-LOX J-2 engines of 200,000 pounds thrust each; and the third stage, designated S-IVB, has a single J-2 engine of 200,000 pounds thrust. Atop the S-IVB stage is the instrument unit which contains instrumentation and launch vehicle guidance. A Saturn vehicle of intermediate size, designated the Saturn IB, is used for earth orbital testing of the Apollo spacecraft. Ground test requirements for the Saturn IB and the earlier Saturn I are similar to those for the Saturn V and are not covered in this paper.

The spacecraft has three major modules: the command module (CM) which carries the three-man crew to lunar orbit and back to earth; the service module (SM) which propels the spacecraft into and out of lunar orbit and provides midcourse corrections to and from the moon; and the lunar excursion module (LEM) which carries two of the three-man crew from lunar orbit to the lunar surface and back to lunar orbit. The

spacecraft engines use the hypergolic propellant combination of hydrazine, unsymmetrical dimethyl hydrazine; and nitrogen tetroxide with pressurized tank feed systems, as contrasted with the pump fed cryogenic propellant launch vehicle engines.

An overall sequence of the Apollo mission is shown in figure 3. The S-IVB stage, the instrument unit, and the spacecraft are first put into earth orbit, and after one to three orbit coast, the S-IVB stage relights and places the spacecraft into a translunar trajectory. During transit to the moon, the LEM is repositioned on the spacecraft and the SM makes midcourse corrections. Upon approaching the moon, the SM decelerates the spacecraft into lunar orbit. The LEM then detaches with two of the three-man crew and descends to the lunar surface. The LEM has two stages, and upon completion of the nominal 24-hour lunar stay, the ascent stage takes off and returns to join the command and service modules (CSM) in lunar orbit. After crew transfer, the LEM is left in lunar orbit and the CSM injects into a transearth trajectory. Before entering the earth's atmosphere, the CM separates from the SM. After decelerating from the initial entry velocity of 36,000 ft/sec to low speed, the CM is landed by means of a three-parachute landing system.

The development and checkout of this space vehicle requires an imposing assemblage of ground test installations. It has been an Apollo Program goal from initiation to configure the ground test program so that no major deficiency will be uncovered in flight that could have been uncovered in ground testing. As a goal, flight test, therefore, will serve to verify or confirm adequacy of the system as developed, qualified, and checked out on the ground.

In the Fall of 1963, an important decision was made by Dr. George E. Mueller, Associate Administrator for Manned Space Flight, when he directed that the Apollo Program implement the All-Up Concept in its flight test program. In this approach, all flight tests from the very first utilize the complete space vehicle in its "all-up" configuration. The previous approach proceeded step-by-step with initial flights of the first stage alone, then flights with the second stage added, and finally, after verification of satisfactory operation of the two stages, flight of the complete vehicle. The revised all-up approach offers the potential for significant time and cost savings and ability to capitalize on success. This approach intensifies reliance on a comprehensive ground test program and the associated ground test installations to ferret out system weaknesses prior to flight.

Paramount in configuring the ground installations for testing the Apollo space vehicle is a consideration of the cost of a flight failure versus the cost of facilities required to provide flight simulation on the ground. As an example, the cost of the space environmental chamber for verifying the capability of the Apollo spacecraft to operate in the space environment is approximately $30 million. If such a facility saved just one Apollo flight test, its cost would be saved several times over. That a comprehensive testing program in such a facility significantly enhances the prospects for mission success, there can be no effective argument. Thus, on economic grounds alone, without regard to the human safety aspect, such a ground installation can be readily justified.

In the following pages, the major development facilities for the launch vehicle and spacecraft will be covered in that order; these discussions will be followed by a treatment of the space vehicle checkout installations.

Launch Vehicle Stage and Engine Test Facilities

A list of the major types of launch vehicle development and qualification ground tests is shown in figure 4, and the phasing of these tests for the Saturn V launch vehicle are shown in figure 5.

Structural tests are conducted on components and structurally complete stages and provide the verification of stage structural integrity. During these tests, the stage is subjected to simulated flight limit loads. Components will be tested to prove the design load carrying capability and establish a margin of safety beyond maximum expected operational environment. Battleship tests are conducted on propulsion stages to investigate overall propulsion system compatibilities and establish system limits. The configuration of these stages will duplicate the flight stage in major respects, with the exception that propellant containers are of heavier construction than the flight article and many flight type systems are not installed. During these tests, repeated engine firings are conducted to evaluate thoroughly the engine-stage performance, propellant feed system operation, and compatibility of major stage systems with engine systems.

All-systems tests are performed on ground test stages of near identical construction to flight stages with all flight type systems installed and with flight type ground support equipment. The objective of these tests is to verify compatibility of all stage, engine, and ground systems in their actual flight configuration. In the case of the S-IC stage, it was decided early in the program to utilize an all-systems type of stage for both battleship and all-systems kinds of testing.

Most development testing of the launch vehicle and the spacecraft is done with individual stages or modules. A major exception is the dynamic testing carried out on the complete space vehicle assembly. Dynamic tests will be conducted on the Apollo-Saturn V space vehicle to determine the dynamic response under critical conditions of flight such as launch, maximum aerodynamic loading, and stage separation. These tests will confirm vehicle flight control system design and verify vehicle structure dynamic analysis. The tests will include evaluation of vehicle bending modes, frequencies, interaction between engine gimbaling motion and vehicle structures, damping characteristics, and local modes and frequencies.

Major ground installations for development testing of the S-IC stage at the NASA Marshall Space Flight Center are shown in figures 6, 7, and 8. The structural testing of the S-IC stage is carried out in the load test facility shown in figure 6. This facility is required to static load the S-IC structure axially to ~13,000,000 pounds. This total load represents the thrust load of five F-1 engines multiplied by the hold-down release amplification factor and by the factor of safety of 1.4. The load is applied hydraulically by multiple cylinders. Concurrently, the flight limit bending load is applied by metal bands attached to a large number of laterally mounted hydraulic cylinders.

Hydrostatic proof tests are carried out in the hydrostatic test stand shown in figure 7. The all-systems static firing tests of the S-IC stage are conducted on the S-IC static test stand pictured in figure 8. This structure is built to withstand the 7,500,000 pounds of thrust of the S-IC stage for its full duration of 150 seconds. Engines are ignited sequentially so that the full 7,500,000 pounds of thrust builds up in a programmed manner. In order to withstand the intense heat of the 30,000 lb/sec of 5,000° exhaust gases, 320,000 gallons per minute of water are pumped to the flame bucket by the 37,500 hp high pressure pumping facility shown in figure 9. Full duration firings of the stage have been successfully completed with negligible damage to the stand. The 336 foot high stand and its foundation required 30,000 cubic yards of concrete and 9,000 tons of steel. One of the major considerations in siting and operation of the stand is the acoustic sound levels produced by the 7,500,000 pounds of thrust. This problem is illustrated in figure 10. Extensive studies of sound propagation in the area as a function of climatic conditions were carried out in arriving at a satisfactory plan. Experience has verified that we were conservative in our siting criteria and noise has not been a restricting factor in the development program.

Production acceptance tests of the S-IC stages will be accomplished on the stands pictured in figure 11 at a remote area in Mississippi designated the Mississippi Test Facility. These test stands are similar in construction to the single MSFC stand but required 98,000 linear feet of steel piling for foundation at the Mississippi site. Here again, acoustics were a major factor in siting the stands and proximity to neighboring cities had to be carefully considered.

The unique and difficult aspect of the S-IC thrust stands construction is simply their physical size and massiveness, representing as they do, a five-fold increase in thrust over previous experience. High tensile structural steel was necessary in stand construction and a number of welding problems arose early in the program. Extensive use of models was made in critical design areas such as the flame bucket shape and cooling water flow rate and distribution. Experience to date has verified the soundness of the design concepts and their implementation.

Let us now turn our attention to the S-II stage. Major stage test phasing is shown in figure 5. Structural testing for the S-II stage is done at Seal Beach, California, and Santa Susana, California. Structural and hydrotest facilities are shown in figures 12 and 13. Here, flight limit axial loads

and bending loads are applied in appropriate combination. While the thrust of the S-II stage is only a fraction of that of the S-IC stage, the physical size of the S-II stage is similar to that of the S-IC because of the low density of liquid hydrogen. Because of the extreme low temperature of liquid hydrogen, coupled with this large physical size, unique structural problems had to be resolved in the design and construction of this stage.

Battleship development testing of the S-II stage is done at the Santa Susana, California, test facility shown in figure 14. A major problem in design and construction of the stand and supporting facilities was provision for safe and economical handling of the very large quantity of liquid hydrogen involved-approximately 300,000 gallons per full duration firing. Full duration firings of 400 seconds duration have been successfully accomplished on the S-II battleship stage at Santa Susana. Prior to doing the full duration firing with all five engines installed, a single engine series of tests was conducted. A photograph of the single engine installation is shown in figure 15. In this picture can be seen the side restraining supports which support the engine during ignition. Subsequent to ignition, explosive bolts are used to free the engine to move normally and to be gimbaled.

The all-systems and acceptance test stands for the S-II stage are located at the Mississippi Test Facility, and the first of the two stands is shown in figure 16. Flight stages will be full duration fired here prior to shipment to Florida for launch. This construction required 145,000 linear feet of piles, 93,000 cubic yards of concrete, and 15,000 tons of steel; the stands are 152 feet high from the base.

The S-IVB stage test phasing is shown in figure 5. In the S-IVB program, because of previous experience with the very similar S-IV stage used in earlier Saturn I, an all-systems type of stage is not used in the development program. The battleship stage has flight type subsystems installed and thus allows verification of the all-systems type, but does not utilize actual flight-weight structure. The S-IVB development program is carried out at Sacramento, California, and the battleship test stand for S-IVB stage development is shown in figure 17. The S-IVB, as well as the S-II, requires facility provisions for full flight duration burning of about 400 seconds. In addition, the S-IVB stage must coast and restart in flight after several hours exposure to the space environment. Thermal conditioning is provided in the S-IVB stand to allow simulation of these relight conditions in space. A second stand at Sacramento is provided for full duration test of flight stages.

Major ground testing for the F-1 and J-2 engines for the S-IC, S-II, and S-IVB stages is done at Edwards Air Force Base and Santa Susana, California, respectively. Major F-1 test facilities are shown in figure 18. Over 1,000 separate tests for a total of 50,000 seconds duration have been run on F-1 engines to date. From a test facility logistics point of view, it is significant to note that while operating, this engine burns approximately 6,000 pounds of propellant per second and that approximately a billion pounds of propellant have been used in developing the F-1 engine to date.

Major J-2 test facilities at Santa Susana are shown in figure 19. Ambient testing of the J-2 engine initially presented a significant problem because of the engine high expansion ratio exhaust nozzle designed for altitude conditions. Severe engine sideloads and buffeting were generated by the over-expanded nozzle and corrective action was necessary. A comparatively simple fix, shown in figure 20, was found to solve the problem. The fix consisted of bolting a short convergent section to the engine exhaust nozzle outlet for all ambient pressure ground testing. The area of contraction is slight and does not interfere with ground test operations. In order to prevent damage due to

asymmetric thrust during starting transients, the engines are restrained laterally during start and then released by firing explosive bolts. Over 1,000 separate tests for a total of over 50,000 seconds duration have been run on J-2 engines to date.

Although much of the J-2 engine testing can be done under ambient sea level conditions, it is essential that an extensive amount of simulated altitude testing be accomplished. For this purpose, a simulated altitude stand is utilized at the Santa Susana test facility. This altitude stand is shown in figure 21. Steam injectors are used to attain initial altitude condition for starting the engine. After starting, the engine assists in sustaining its own altitude exhaust condition. An exhaust pressure equivalent to an altitude of about 100,000 feet is maintained.

In the third stage S-IVB application, the J-2 engine is subjected to a long duration coast in the space environment prior to restart. Thus, a thermal soak, cold start condition exists for the J-2 which is not the case for the F-1 engine. In order to investigate the behavior of the J-2 engine in its altitude start condition, a thermal conditioning test set-up will be used in which the engine will be subjected to simulated solar heating on one side and low temperature space environment

on the other, for a time duplicating the in-space coast phase. These tests will be accomplished in an altitude test facility at the Arnold Engineering Development Center at Tullahoma, Tennessee.

Space Vehicle Dynamic Testing

The primary purpose of dynamic testing of space vehicles is to verify theoretical and analytical models which define the dynamic characteristics of the vehicle system. For these tests, the complete space vehicle--launch vehicle and spacecraft--is used. A detailed knowledge of mode shapes and frequencies is essential in control system design. The control system must differentiate between bending of the vehicle and rigid body motions in order to control attitude and direction, as well as to stabilize the coupled oscillations of the vehicle structure, engines, and propellants. As shown in figure 22, the control system is designed initially on the basis of a theoretical analysis of vehicle dynamics. This is followed by an analog simulation study of the vehicle and its control system, based on theoretical analyses. Because of limitations in available theory and configuration arrangements and details which are not amenable to theoretical analysis, such as the effect of joint flexibility, it is necessary to verify experimentally the dynamic behavior of the vehicle. For full

scale dynamic tests the launch vehicle is suspended in a spring suspension system with support only at the bottom near the engine gimbal plane. With the vehicle thus freely suspended, electromagnetic shakers are used to provide lateral and longitudinal vibration inputs. Lateral excitation is applied on the pitch axis and then the yaw axis to check differences in characteristics in these two perpendicular planes. In addition, testing is conducted with roll or torsion excitation by placing a shaker against each of two diametrically opposed fins and exciting the vehicle in a circumferential direction.

Longitudinal excitation can be applied simply by placing shakers under the vehicle and exciting it vertically. The suspension system must be capable of supporting the vehicle adequately for any of these directions of excitation. Figure 23 shows the full scale Saturn I dynamic test vehicle being positioned in the dynamic test tower. The drawing in figure 24 shows somewhat better some of the details of the dynamic test article. The heavy beam shown in cutaway supports the weight of the vehicle during the dynamic test. From this beam hangs the suspension system which attaches to the vehicle at the bottom near the gimbal plane. The suspension system consists of eight sets of multiple springs, each attached to

multiple cable sets which, in turn, are connected at the bottom to clamps which hold the vehicle. The stiffness of the springs affects the results of the test and the suspension should be kept as soft as possible for any given configuration to minimize this effect. Both the number of spring cable sets used and the number of springs in each package can be varied to achieve best results. Since the vehicle is supported only at the bottom, it is free to rock and could be damaged by contact with the tower. To prevent such damage, impact snubbers are provided at the top of the first stage. The system is such that the snubbers are in contact only rarely, and usually only when winds become excessive. Supports are provided onto which the vehicle is lowered between tests or when wind difficulties arise. Also shown in figure 24 is a typical shaker in position for lateral vibration of the vehicle. The shakers are electromagnetic and provide a sine wave force input. The force input is at the gimbal plane. This single location was found to be adequate, if not optimum, during early studies of the effects of shaker location. It has the added advantage for control system work of locating the force input where engine gimbal forces act on the vehicle. This allows direct determination of transfer functions required for control system design from the results of the dynamic test. These transfer functions are in the form of flight sensor outputs at various points on the

vehicle as a function of force input at the gimbal plane. Such functions are required in order to filter out vehicle vibration from the control system loop.

The full scale stand for the larger Saturn V dynamic testing is shown in figure 25. An improved hydro-dynamic suspension system is now under development for dynamic testing of the Saturn V. A schematic of this system, together with a sketch of the Saturn V in the tower, is shown in figure 26. Four hydro-dynamic support units, each atop a steel and concrete pedestal, support the vehicle at the four hold-down points. This is the optimum location for supporting the vehicle load. This system is estimated to cost considerably less than a new cable-spring system for this larger vehicle. The system consists of a cylinder fixed to the pedestal, with an internal piston which is free to move vertically, but which is supported laterally by a high pressure oil gap which prevents contact with the cylinder. The vertical spring rate is determined by the trapped volume of air which can be varied easily to change the spring rate over a wide range. Lateral motion of the vehicle is provided for by a horizontal oil bearing on top of the piston, and rotational motion is provided by a spherical bearing above the horizontal bearing. This system has a number of important advantages over the cable-spring system. It is simple to analyze, the effective

mass attached to the vehicle is much lower than that for the cable-spring system; and damping of the system is extremely low (well under 100 pounds of damping force per support). This system is highly satisfactory for longitudinal testing, as well as for lateral and torsional testing. The diameter of the piston is approximately 30 inches and the overall height of the system, not including the pedestal, is about 90 inches. One of the big advantages of this system is ease of handling the vehicle. The vehicle can be picked up merely by pressurizing the supports and set down simply by depressurizing. Spring rate changes require only pressure changes which can be accomplished quickly without mechanical changes. Safety advantages also exist, particularly in that failure of one support hydraulic system does not damage the vehicle because the other supports take up the load.

In addition to the full scale dynamic tests of the Saturn vehicles, an extensive program of model dynamic testing is underway. The primary advantage to be gained from model testing is the reduction in the number of full scale tests which are required. Model tests, which are far less expensive, and can be run much earlier, identify the configuration and narrow the conditions which need to be investigated in full scale. In addition to helping plan the full scale test

program, the model tests give a preliminary verification of theory, and give an early indication of the optimum location of instrumentation, both for ground test and for flight.

Two of the scale models undergoing dynamic testing at the NASA Langley Research Center are shown in figure 27. The larger is a 1/10 scale model, roughly 35 feet high, and the smaller is a 1/40 scale model which is about 9 feet high. A series of the 1/40 scale model tests has been completed, and the testing of the 1/10 scale model is well along.

Spacecraft Ground Test Facilities

In addition to withstanding the launch environment, the spacecraft must undergo the environmental extremes of long duration soak in the extreme thermal-vacuum conditions of space, lunar surface conditions, lunar landing, atmospheric reentry, and earth landing. Each module of the spacecraft has its own peculiar test requirements, as does the integrated spacecraft. These requirements are listed in figure 28. The command module is the only part of the spacecraft which returns to earth and therefore, it alone requires reentry and earth landing test facilities. Land and water impact tests are required to simulate the conditions of return to earth for a water or a land landing under various combinations of horizontal

and vertical velocity and impact direction and angle. The tower shown in figure 29 is used for these tests. This facility, located at the North American Aviation factory in Downey, California, consists of an open steel double "A" frame tower structure with connecting buttress 143 feet high and 143 feet long, with a water pool at one end and an earth impact area at the other. The pool is 90 feet long, 40 feet wide, and 20 feet deep. A variable height parallelogram suspension is provided. By varying the height of drop and angle of swing, all necessary combinations of horizontal and vertical impact velocity can be attained. Nominal impact conditions are approximately 23 feet per second vertical, and 35 feet per second horizontal velocity for a landing on water. Emergency conditions extend vertical velocities of impact to 30 feet per second in conjunction with horizontal velocities of over 40 feet per second on land. Over 100 thoroughly instrumented drop tests have been made from this tower over the complete range of impact conditions. Anthromorphic dummies are carried during some of these tests to verify acceptable levels of impact deceleration and adequacy of seat shock attenuation.

Reentry heating tests are accomplished on scale models in arc heated jets and with full scale quartz lamp heating installations such as shown in figure 30. The scale model

tests are used to determine heat flux distribution and ablator effectiveness under simulated reentry conditions. The full scale heat lamp tests are used to verify adequacy of the insulation and load carrying structure under the long duration, high temperature conditions associated with manned reentry at lunar return velocity. In these tests heat flux is automatically programmed to vary realistically with time during the 2,000 second reentry heating period. Full scale reentry heating to the external ablative structure cannot be duplicated in ground installations and verification by flight test is required.

Ascent and reentry acceleration loads on the command module and the three-man crew are simulated in full scale on a flight acceleration facility, or large centrifuge, located at the Manned Spacecraft Center in Houston, Texas. This facility, shown in figures 31 and 32, is used to train the three-man spacecraft crew, to develop their equipment under acceleration stresses, to evaluate the physiology of the men under space flight stresses, and to evaluate human tolerance to specified acceleration profiles. The facility consists basically of a large rotating, enclosed truss arm supporting a spherical gondola. The environment within the gondola is temperature, pressure, and humidity controlled to simulate spacecraft conditions. The arm is rotated at speeds up to

42 rpm by a single directly coupled motor rated at a maximum of 10,700 hp and capable of delivering torques of up to 5.1 million pound feet. The drive system is controlled by a digital computer which controls the revolutions per minute of the drive rotor to give the desired acceleration versus time profile. The centrifuge arm is of welded stainless steel truss construction and can support an acceleration onset rate of 5-8 g's per second when operating at a radial acceleration of 2 to 30 g's. The facility will be able to simulate an acceleration of 20 g's for extended periods of time, and will be able to test at 30 g's for up to about three minutes.

The gondola is approximately spherical in shape, with a diameter of about 144 inches. The gondola is gimbaled and is oriented in one axis by a 150 hp electro-hydraulic servo motor. A maximum angular acceleration of 10 radians per second per second, and a maximum angular velocity of 3.1 radians per second (30 rpm) are possible. The second axis of rotation of the gimbaled gondola is driven by a 100 hp electro-hydraulic servo motor which is capable of producing an angular acceleration of 2 radians per second per second, and an angular velocity of 3.1 radians per second (30 rpm). This performance is obtained with a design payload of 3,000 pounds located within the gondola. This payload includes personnel, life support equipment, crew equipment, and all instrumentation

necessary to collect and transmit the required data and control information from the gondola. The total arm end load, including the payload, the gimbal ring, and the gondola, is about 24,000 pounds. Gondola internal environment can be controlled within a temperature range from 50°F to 200°F, a relative humidity of 40 to 60%, and pressure down to 5 psia. The facility may be controlled and operated in automatic, semi-automatic, or manual modes. In the automatic mode, the test is pre-programmed and may be controlled and operated by the computer. Additionally, in the automatic mode the test program may be controlled and operated by the computer, but the program can be modified by test subject input signals within pre-programmed limits. In the semi-automatic mode, the centrifuge operator controls the speed of the arm, while the other functions are computer controlled. In the manual mode, the test program may be controlled manually by the centrifuge operator, with the computer operating with pre-programmed safety values to limit operations which would damage the machine or test subjects. Lastly, the facility can be operated manually without the computer in operation. With this facility will exist, for the first time, the capability to simulate ascent and reentry acceleration profiles for multiple manned crews under realistic spacecraft environmental conditions. Actual Apollo command module interior cabin and control panel arrangements will be installed in the gondola.

Perhaps the most sophisticated ground test facility required in development and qualification of the Apollo space vehicle is the space environment simulation laboratory located at the Manned Spacecraft Center in Houston, Texas. The function of this laboratory is to provide simulated space thermal-vacuum testing of Apollo and future spacecraft systems, and to evaluate the capability of man to perform useful functions inside and outside spacecraft on a simulated lunar surface. An artist's sketch of the space environment simulation laboratory is shown in figure 33. The larger of the two chambers provided, Chamber A, is further illustrated in figure 34. This larger chamber is intended primarily for combined tests involving men and operating Apollo spacecraft. The smaller chamber, Chamber B, to be discussed subsequently, will be utilized for life systems and astronaut training studies, in addition to tests of single modules of the Apollo spacecraft.

Chamber A is a 65-foot diameter, stainless steel vessel, having an overall height of 120 feet. The chamber can accommodate spacecraft up to 65 feet in height and 25 feet in diameter. A special feature is the rotating platform, or lunar plane, which is built into the floor of the test chamber. This platform is 45 feet in diameter and can support a spacecraft weighing as much as 150,000 pounds. The platform can be rotated $\pm$ 180 degrees at speeds up to one and two thirds revolutions per

minute under either manual or automatic control to simulate the rotation of the spacecraft relative to the sun. The temperature of the platform can be controlled to simulate lunar surface temperatures in the range of 80° to 400° Kelvin. The chamber is equipped with a hydraulically opened and closed 40-foot diameter door located in the cylindrical wall of the chamber; a photograph of the door during construction is shown in figure 35. This chamber uses 14 diffusion pumps, 32 inches in diameter, with 50,000 liters per second pumping capacity. Both chambers have black liquid nitrogen cooled heat sink panels which will operate at approximately 80° Kelvin. Insofar as feasible, all surfaces in the chamber viewed by the vehicle consist of heat sink panels. Chamber A is also equipped with 1,300 square feet of cryo panels cooled to $14\frac{1}{2}$ to $20\frac{1}{2}^{\circ}$ Kelvin by gaseous helium. Chamber A is designed to operate at a pressure of 1×10^{-5} torr, with an estimated maximum outgassing load of 27.6 torr/liter/second. Pumpdown time will be approximately 19 hours.

Top and side wall solar simulators of the carbon arc type with a wave length range from approximately .25 to 3.0 microns are fitted. The target of the side sun is 13 feet wide by 33 feet high, with provision for expansion to 20 feet wide by 65 feet high. The target area of the top sun is 13 feet in

diameter with provision for expansion to a diameter of 20 feet. Radiation intensity can be controlled in the range from 60 to 135 watts per square foot.

The smaller chamber, shown in figure 36, is a 35-foot diameter stainless steel vessel with an overall height of 43 feet. Vehicles up to 16 feet in diameter and 27 feet in length can be accommodated. Access is by a removable top. Spacecraft of up to 75,000 pounds weight can be supported on a fixed lunar plane 20 feet in diameter. The surface temperature of the lunar plane can be controlled in the same manner as for Chamber A. The vacuum roughing system for both chambers, A and B, consists of two mechanical vacuum booster pumps and six oil sealed rotary vacuum pumps split into trains of three each. Backing systems for each of the diffusion pumping systems consist of two parallel trains which are used during pumpdown. Chamber B has twelve diffusion pumps, 32 inches in diameter, with 50,000 liters per second pumping capacity. Chamber B is designed to operate at a pressure of 1×10^{-4} torr, with a maximum outgassing load of 25.6 torr/liter/second. The solar simulators for Chamber B are of the same type as for Chamber A, but only a top sun is fitted. The target area for the top sun is 5.6 feet in diameter with provision for expansion to a diameter of 20 feet.

Unique features are provided for both chambers as a result of their requirement for manned operation. Chamber A is fitted with an emergency repressurization system which will raise the total pressure to 6 psia and the oxygen partial pressure to 4 psia, respectively, within 30 seconds. The emergency repressurization is accomplished by admitting a small quantity of nitrogen into the chambers to blanket the hot oil in the diffusion pumps until the pump valves can be closed. This is done to reduce the hazard of an explosion or fire when oxygen enters the chamber. Oxygen is then admitted until the partial pressure of the oxygen reaches 4 psia. Under normal conditions, about 30 minutes are required for the chamber pressure to rise to 6 psia, or three hours to repressurize to sea level conditions. Filtered atmospheric air is the normal repressurizing medium.

Man locks will be fitted to both chambers. Chamber A will have two man locks; one double lock at the lunar plane, and one single lock at the mid-chamber level. Man lock doors are side-hinged and provided with quick action clamp devices for initial seal. Provisions are made for pressure-suited personnel to be directly connected to two umbilical lines to external environmental control systems.

The space environment simulation laboratory test operation concept allows for independent spacecraft tests to be conducted in Chambers A and B concurrently. Therefore, various combinations of the Apollo spacecraft modules can be tested without stopping tests on the others. It is possible to test all of the Apollo spacecraft modules together in Chamber A, thus allowing time in Chamber B for other tasks, such as the development of advanced space suits. Initial plans are to test command and service modules as one spacecraft in Chamber A, and the ascent and descent stages for the lunar excursion module in Chamber B. With the exception of the propulsion, electrical power generation, and environmental control subsystems, all systems aboard the spacecraft will be able to operate in normal mode during simulated lunar missions in the environmental chambers. Although hot firings of the spacecraft propulsion systems are not feasible because of safety and operational considerations, simulated propellants having thermal properties closely resembling the actual propellants will be utilized during tests to establish realistic temperature profiles and histories within the spacecraft.

Spacecraft Vibration and Acoustics Test Facility

A schematic of the Manned Spacecraft Center spacecraft vibration laboratory is shown in figure 37. This laboratory has the capability for vibration testing of an integrated Apollo spacecraft in its launch attitude. Two specific types of vibration testing will be performed: (a) environmental testing during which the spacecraft will be subjected to high vibratory energy levels through the use of a thruster system consisting of six 10,000 pound force thrusters; (b) modal testing wherein the spacecraft will be subjected to low vibratory energy levels induced by a modal system consisting of six 150 pound force shakers. The energy level will be high enough to excite the structure for determination of resonant frequencies, mode shapes, and damping characteristics. The spacecraft vibration laboratory is 60 feet square, with a clear height of 93 feet and contains a suspension system which will allow vibration tests on an integrated spacecraft suspended vertically. A spacecraft up to 30 feet in diameter and weighing 150,000 pounds can be accommodated. The 10,000 pound force thrusters differ from shakers in that they use a one and one half inch rod as a driver instead of having a table top armature. The thrusters are designed to be self isolating and require no seismic mass to react the shaker force. Three 175 kilowatt power amplifiers are located in

an adjacent room. Each power amplifier is used to drive two thrusters. The thruster system contains a sine control console and an automatic random control console which allow various methods of control for the thruster units, including feedback from the response of the spacecraft. The modal system, with its six 150 force pound shakers, six one kilowatt power amplifiers, and a sine control console, is portable and can be operated by one man.

The spacecraft acoustic laboratory at the Manned Spacecraft Center is shown in figure 38. This laboratory is designed primarily for fluctuating pressure qualification testing of manned spacecraft in the launch configuration and orientation. The 60-foot by 70-foot by 105-foot high structure will accommodate a spacecraft up to 30 feet in diameter and 85 feet high.

The primary test technique consists of bathing the entire exposed surface of the spacecraft with controlled high intensity acoustic noise in the form of sixteen separate progressive waves. These progressive waves are directed downward over the vehicle in sixteen separate ducts or shrouds which span the spacecraft circumferentially. The shape of the ducts is adjustable so that sound pressure levels and gradients up to a maximum overall sound pressure level of 171 decibels (.0002 dynes per square centimeter) can be achieved longitudinally. The sixteen

noisemakers, which are 10,000 acoustic watt, electrically programmed, high intensity air modulators, are suspended from the ceiling of the tower and attached to 30 cycle per second cut-off horns connected to the shrouds. Each noisemaker is independently controllable from the control room so that the acoustic circumferential distribution can be point programmed. The 160,000 acoustic watts of power will provide an overall sound pressure level of 171 decibels at the horn-shroud interface. In a full reverberant mode of operation, this amount of acoustic power will create a sound pressure level of approximately 157 decibels in the tower. The large volume of air, approximately 25,000 cubic feet per minute at 40 psig, required by the noisemakers is supplied by an air compressor located in an adjacent building.

Spacecraft Propulsion Testing

The Apollo spacecraft propulsion system uses highly toxic hypergolic propellant combinations of unsymmetrical dimethylhydrazine, hydrazine, and nitrogen tetroxide. For that reason, major development test facilities were located in a remote area--the White Sands Missile Range in New Mexico. It was desired to have the capability to make full flight duration runs with flight-weight propulsion systems with actual flight coast and relight cycles. This requires assurance of freedom from interruption for periods up to lunar mission duration of

approximately seven days. The integrated propulsion development facility, constructed for this purpose at White Sands, New Mexico, is shown in figures 39 and 40. Included are five test stands for the ascent and descent stages of the lunar excursion module and the service module propulsion system. The two service module test stands have ambient pressure exhaust systems, while two of the three lunar excursion module stands provide altitude capability. In the case of the service module, a detachable exhaust nozzle may be removed for the majority of ground testing activity, and development tests for the service module at White Sands are run in that configuration. In the case of the lunar excursion module descent engine, when throttled in its minimum thrust condition of about 10% of full thrust, the chamber pressure is below ambient pressure. For operation at this condition an altitude exhaust capability obviously is required. In addition, the exhaust nozzle for the lunar excursion module descent engine is not readily removable and, therefore, the engine is run in its full space configuration. Altitude exhaust conditions for the lunar excursion module descent engine are produced by steam driven diffuser-ejector systems. The ejectors are operated by alcohol-oxygen burners with downstream water injection. In order to support the 10,000 pound thrust LEM descent engine, an ejector flow rate of approximately 560 lbs/sec is required.

For altitude operation the complete LEM stage is encapsulated within a pressure-tight vessel which is connected to the ejector system. During operation a simulated altitude of approximately 100,000 to 150,000 feet, depending on operating conditions, is sustained about the vehicle and at the engine nozzle exit. Although the service module can be tested at sea level conditions by removing the exhaust nozzle, it is essential also to perform qualification testing of the complete system with the nozzle installed under altitude conditions. This testing is done at the Arnold Engineering Development Center facility of the U.S. Air Force at Tullahoma, Tennessee, where a facility similar to that described for the lunar excursion module is available to accommodate the 22,000 pound thrust service module engine. Tests there are conducted with heavyweight propellant tankage to minimize the risk associated with hypergolic, toxic propellants.

Langley Lunar Landing Research Facility

Early in the lunar landing program it was considered desirable to provide a full scale lunar landing simulation facility where the terminal phase of the LEM landing could be investigated experimentally. Figure 41 shows this recently completed facility which is located at the NASA Langley Research Center at Langley Field, Virginia. This facility provides for

an automatic servo-controlled tether which relieves 5/6 of the weight of a simulated lunar landing vehicle. The tethering cable is attached to the vehicle through a set of gimbals giving rotational freedom. In addition to automatic servo control of tension in the cable which gives vertical freedom of motion, automatic control is also provided for position of a traveling bridge crane and of a dolly mounted on the bridge crane so that both cross range and down range freedom of motion are attained as well, thus giving six degrees of freedom. The tether attached vehicle provides characteristics similar to the LEM, providing for a crew of two, and a controllable thrust landing rocket engine. The vehicle can be maneuvered within a space about 400 feet long, 50 feet wide, and 180 feet high at speeds up to 25 feet per second.

Initial tests with this facility have verified the capability to simulate typical terminal phase lunar landing design conditions. As well as providing important research and design verification data, it is expected that this facility will provide valuable training to the lunar landing crews, particularly because of the unusual angle of approach of the lunar landing vehicle as compared with helicopters or aircraft experience on earth. This program is carried out by the Langley Research Center under the direction of the NASA Office of Advanced Research and Technology and is typical of a number of applied research programs carried out by the several NASA research centers in support of the Apollo Program.

Apollo Spacecraft and Saturn Launch Vehicle Checkout-Installations

Checkout as used in this paper means the determination of the readiness of a subsystem or system to perform its intended function. Checkout of the launch vehicle and spacecraft is a continual process, as the spacecraft modules and launch vehicle stages progress through factory, remote test site, and pre-launch testing. The checkout of the Apollo system utilizes approximately 50 automated checkout stations containing over 100 general purpose digital computers. Approximately 90 computers are used, in addition, for performing data reduction and post-checkout analyses of the data developed during checkout.

Shown in figure 42 is a simplified block diagram of the checkout flow. At Downey, California, North American Aviation assembles and performs factory checkout of the CSM, while at Bethpage, New York, Grumman Aircraft Engineering Corporation assembles and performs factory checkout of the LEM. After these factory tests, the spacecraft modules are shipped to KSC where they are assembled and checked out as an integrated spacecraft before mating with the launch vehicle in the Vertical Assembly Building in the launch area. The production S-IC stages are assembled and checked out by the Boeing Company at the factory in Michoud, Louisiana, sent to the Mississippi

Test Facility (MTF) for static tests, and back to Michoud for refurbishing and post-static checkout. Early S-IC stages undergo a similar process, but are built and tested at the Marshall Space Flight Center in Huntsville, Alabama. The S-II stage is assembled and checked out by North American Aviation at Seal Beach, California, and sent to MTF for static firing and post-static checkout. The S-IVB stage is assembled and checked out by Douglas Aircraft at Huntington Beach, California, and sent to Sacramento, California, for static firing and post-static checkout. The instrument unit is assembled and checked out at the International Business Machines factory in Huntsville, Alabama. After completion of post-static checkout, all stages are shipped to KSC for assembly, mating, and checkout as an integrated launch vehicle. At KSC the complete launch vehicle and spacecraft are assembled into an integrated space vehicle and checked out in the Vertical Assembly Building prior to being transported to the pad for final countdown and launch.

Figure 43 shows a simplified block diagram of the spacecraft checkout system. Two of these systems are used to independently check out the CSM and the LEM. About 1,100 measurements are continuously monitored on the CSM, and about 1,400 measurements are continuously monitored on the LEM. A photograph of one of the spacecraft automatic checkout stations at North American Aviation in Downey, Californis, is shown in figure 44.

Start modules provide a means for the test engineers to command systems within the command and service modules. As these tests are conducted, the results are displayed on event indicators and recorders, analog meters and recorders, and on alphanumeric cathode ray tube displays. This command and display system permits the test engineer to make a determination of the flight readiness of the system. Such library information as permissible tolerances can be furnished the test engineer through the alphanumeric display. Commands entered in the start modules by the test engineer are interpreted by the command computer system. These are sent through a video hardline link to a system which applies the stimuli to the CSM. Carry-on equipment aboard the CSM sample the test points and digitize the data. These data are transmitted through a video hardline link to a digital acquisition and decommutation system and are either directly displayed to the test engineer or processed by the display computer and presented on an alphanumeric cathode ray tube display system to the test engineer.

Shown in figure 45 is a simplified block diagram of the Saturn V launch vehicle checkout system. This system continuously samples approximately 2,550 test points. The launch vehicle checkout system consists of the same basic components as the spacecraft; the central computer, control and storage, the signal conditioning equipment at the vehicle, and the displays and command system for man-machine interface.

Figure 46 presents a simplified block diagram of the integrated space vehicle checkout system. Data flow between the spacecraft and launch vehicle checkout systems is provided through hardline data links. Voice links are also provided between the test engineers and the test conductors situated at the control consoles. The systems are kept in step by means of a reference time base, real time as well as countdown time, and by means of a detailed countdown procedure which is followed by the test engineers.

As of this date, the Apollo Saturn automatic checkout system has been proven by breadboard testing in its overall system configuration, and checkout stations for the CSM, S-IVB stage, and instrument unit are in use for factory checkout of articles scheduled to fly in earth orbit on the Saturn IB launch vehicle in 1966. First utilization of the complete integrated system will occur in late 1966 in preparing for Saturn V launchings in 1967.

CONCLUDING REMARKS

The Apollo Program is the largest research and development program ever undertaken by the United States of America. It entails the effort of more than 250,000 people and will entail a total investment of approximately $20 billion. A significant part of the total effort is related to provision and use of a wide range of ground installations for development, qualification, and checkout of the Apollo flight system. As noted at the outset, only selected major ground installations have been discussed in this paper. The total scope of our ground installations investment is considerably greater than discussed here. There are, in addition, the myriads of installations required for testing the individual vehicle subsystems--such as the environmental control systems, the hydrogen-oxygen fuel cells, the guidance systems, the communications systems, and a host of other intricate systems, each requiring its unique kind of ground test installation. Furthermore, the launch site installations are a subject warranting full discussion as a separate topic. The total brick and mortar investment alone in Apollo ground installations approximates $1.5 billion. It is only through the investment in such ground test installations that we can have reasonable confidence in early success with our flight program. The recent completion of our Saturn I program with the successful launch of all ten vehicles provides significant evidence of the benefits to be derived from a comprehensive ground test program and the requisite ground installations.

The ground installations provided through the investment in the Apollo Program make an excellent base on which we can build future manned space flight development programs. As we move beyond the initial lunar landings to the applications of Apollo hardware for other missions in earth orbit, in lunar orbit, and on the lunar surface, we expect to use these installations repeatedly in the years to come. And beyond these programs, these installations have substantial growth capability to accommodate systems larger than used in the Apollo Program.

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LIST OF ILLUSTRATIONS

FIGURE	TITLE
1	Apollo Test Program Phasing
2	Saturn V - Apollo
3	Apollo Mission Profile
4	Apollo Launch Vehicle Ground Tests
5	Saturn V Major Stage Ground Test Program
6	Structural Qualification Test of S-IC-S Intertank Assembly
7	Moving S-IC Forward Bulkhead Into Hydrostatic Test Stand
8	S-IC-T Being Installed in Development Test Stand
9	Interior High Pressure Water Facility - Huntsville
10	Possible Effects of S-IC Test Firing on Community Property
11	S-IC Test Stand - MTF
12	S-II Structural Test Facility - Seal Beach
13	S-II Vertical Assembly and Hydrotest Building - Seal Beach
14	S-II Battleship Test Facility - Santa Susana, California
15	Underside View of J-2 Test Engine in Battleship Test Stand
16	S-II Test Stand A-2 - MTF
17	S-IVB Battleship Firing (415 seconds) Sacramento
18	F-1 Engine Test Facilities
19	J-2 Test Facilities
20	J-2 Engine Firing With Hold Down Arms; J-2 Engine Mounted Compensator
21	J-2 Engine Altitude Test Facility
22	Data Flow Chart for Design of Saturn V Vehicle Control System
23	Saturn I Vehicle in Dynamic Test Stand
24	Details of Saturn I Vehicle in Dynamic Test Stand

FIGURE	TITLE
25	Saturn Dynamic Test Stands - Huntsville
26	Improved Support System for Saturn V Dynamic Test
27	Scale Model of Saturn V Vehicle
28	Spacecraft Development Ground Test Requirements
29	Command Module Boilerplate Ground Test - Water Impact Test NAA, Downey - Late 1963
30	Apollo Command Module Thermal Test Installation
31	Flight Acceleration Facility
32	Centrifuge System
33	Space Environment Simulation Laboratory
34	Space Environment Simulation Laboratory - Chamber A
35	Space Environment Simulation Laboratory - Chamber A Door
36	Space Environment Simulation Laboratory - Chamber B
37	MSC Vibration Test Facility - Spacecraft Vibration Laboratory
38	MSC Acoustic Test Facility - Spacecraft Acoustic Laboratory
39A	Service Module Test Stands - White Sands
39B	Lunar Excursion Module Test Stands - White Sands
40	Lunar Excursion Module Propulsion System Development Facility
41	Lunar Landing Research Facility
42	Apollo Spacecraft and Saturn V Launch Vehicle Checkout
43	Apollo Spacecraft Automatic Checkout System
44	ACE - Spacecraft Control Room
45	Saturn V Launch Vehicle Automatic Checkout System
46	Apollo/Saturn V - Space Vehicle Checkout System

APOLLO TEST PROGRAM PHASING

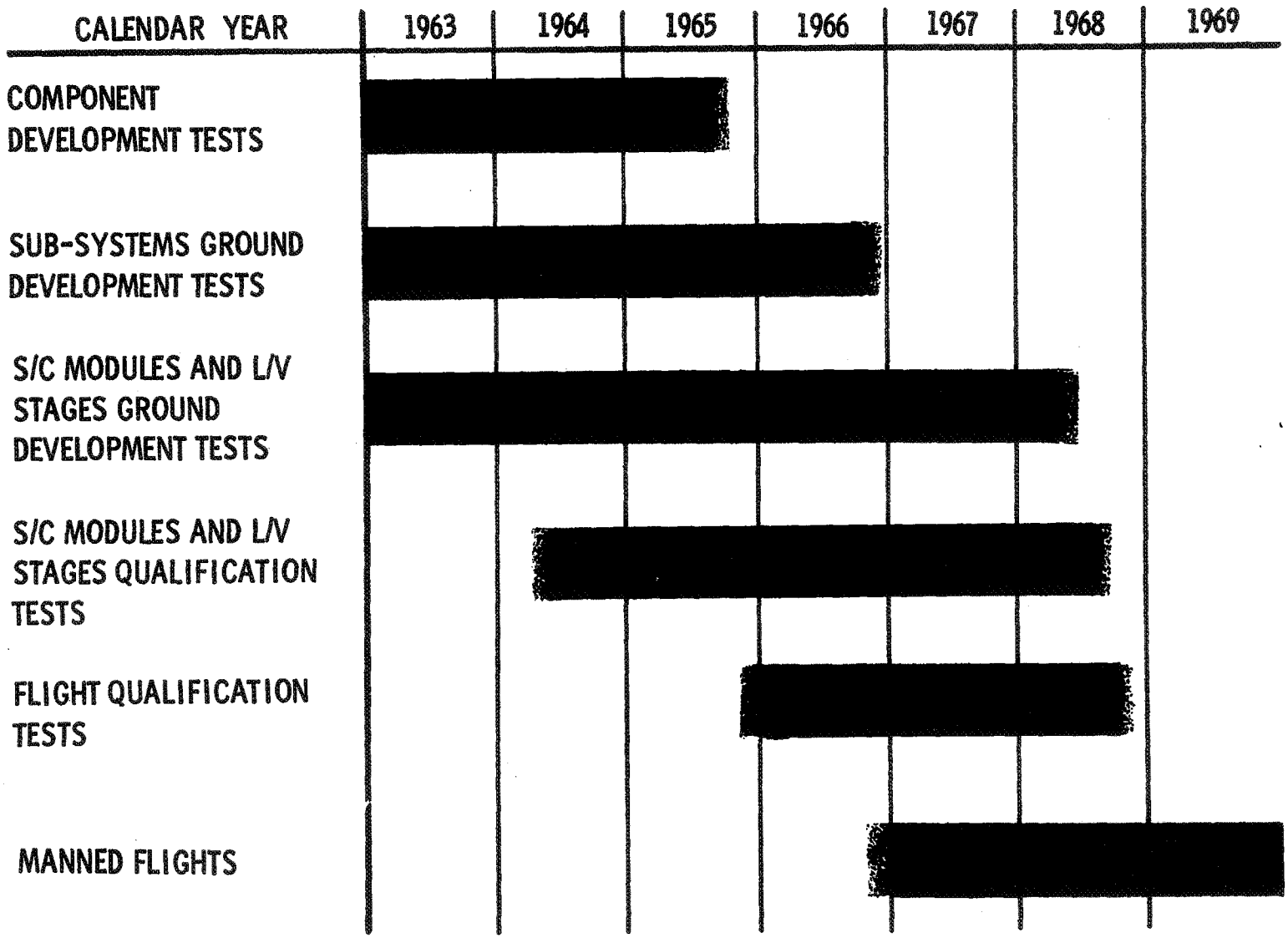

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FIGURE 1 REV. 8/26/65

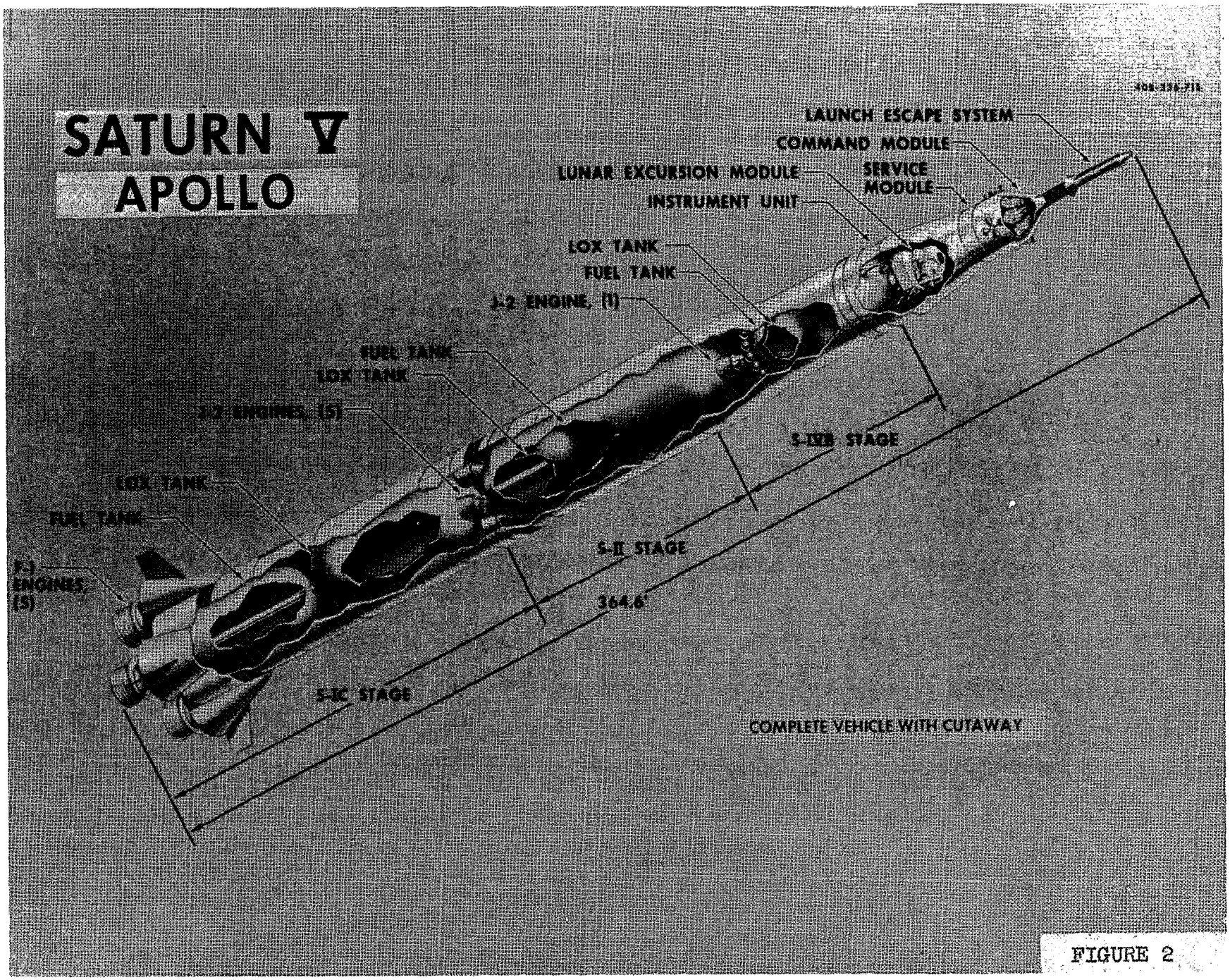
SATURN V
APOLLO
LAUNCH ESCAPE SYSTEM
COMMAND MODULE
LUNAR EXCURSION MODULE
SERVICE MODULE
INSTRUMENT UNIT
LOX TANK
FUEL TANK
J-2 ENGINE, (1)
FUEL TANK
LOX TANK
J-2 ENGINES, (5)
S-IVB STAGE
LOX TANK
FUEL TANK
S-II STAGE
F-1 ENGINES, (5)
364.6'
S-IC STAGE
COMPLETE VEHICLE WITH CUTAWAY

FIGURE 2

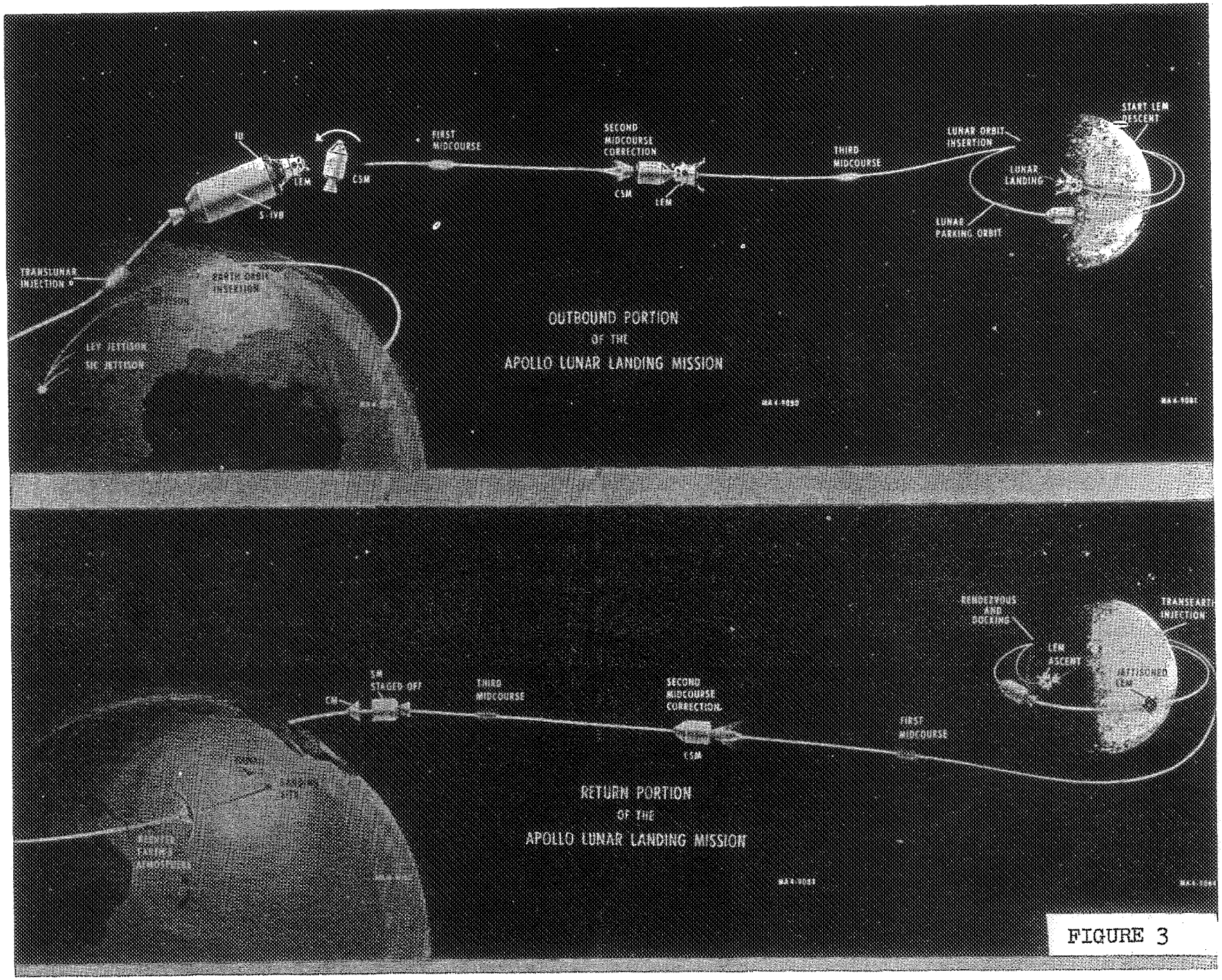
IU
LEM
CSM
S-IVB
FIRST MIDCOURSE
SECOND MIDCOURSE CORRECTION
CSM
LEM
THIRD MIDCOURSE
LUNAR ORBIT INSERTION
START LEM DESCENT
LUNAR LANDING
LUNAR PARKING ORBIT
TRANSLUNAR INJECTION
EARTH ORBIT INSERTION
OUTBOUND PORTION
OF THE
APOLLO LUNAR LANDING MISSION
RENDEZVOUS AND DOCKING
LEM ASCENT
JETTISONED LEM
CSM
LEM STAGED OFF
THIRD MIDCOURSE
SECOND MIDCOURSE CORRECTION
CSM
FIRST MIDCOURSE
RETURN PORTION
OF THE
APOLLO LUNAR LANDING MISSION

FIGURE 3

APOLLO LAUNCH VEHICLE GROUND TESTS

TYPES OF TESTS	OBJECTIVES OF TEST
STRUCTURAL	VERIFY STRUCTURAL INTEGRITY
BATTLESHIP	PROPULSION SYSTEM TESTS (HEAVY WEIGHT TANKS)
DYNAMIC	SIMULATION OF FLIGHT DYNAMICS ENVIRONMENT ON STRUCTURE (NON-PROPULSIVE FLIGHT CONFIGURATION STAGE)
ALL-SYSTEMS	QUALIFICATION OF ALL STAGE SYSTEMS (FLIGHT HARDWARE AND LAUNCH TYPE GSE)

NASA M64-169
REV. 8/26/65

FIGURE 4

SATURN V MAJOR STAGE GROUND TEST PROGRAM

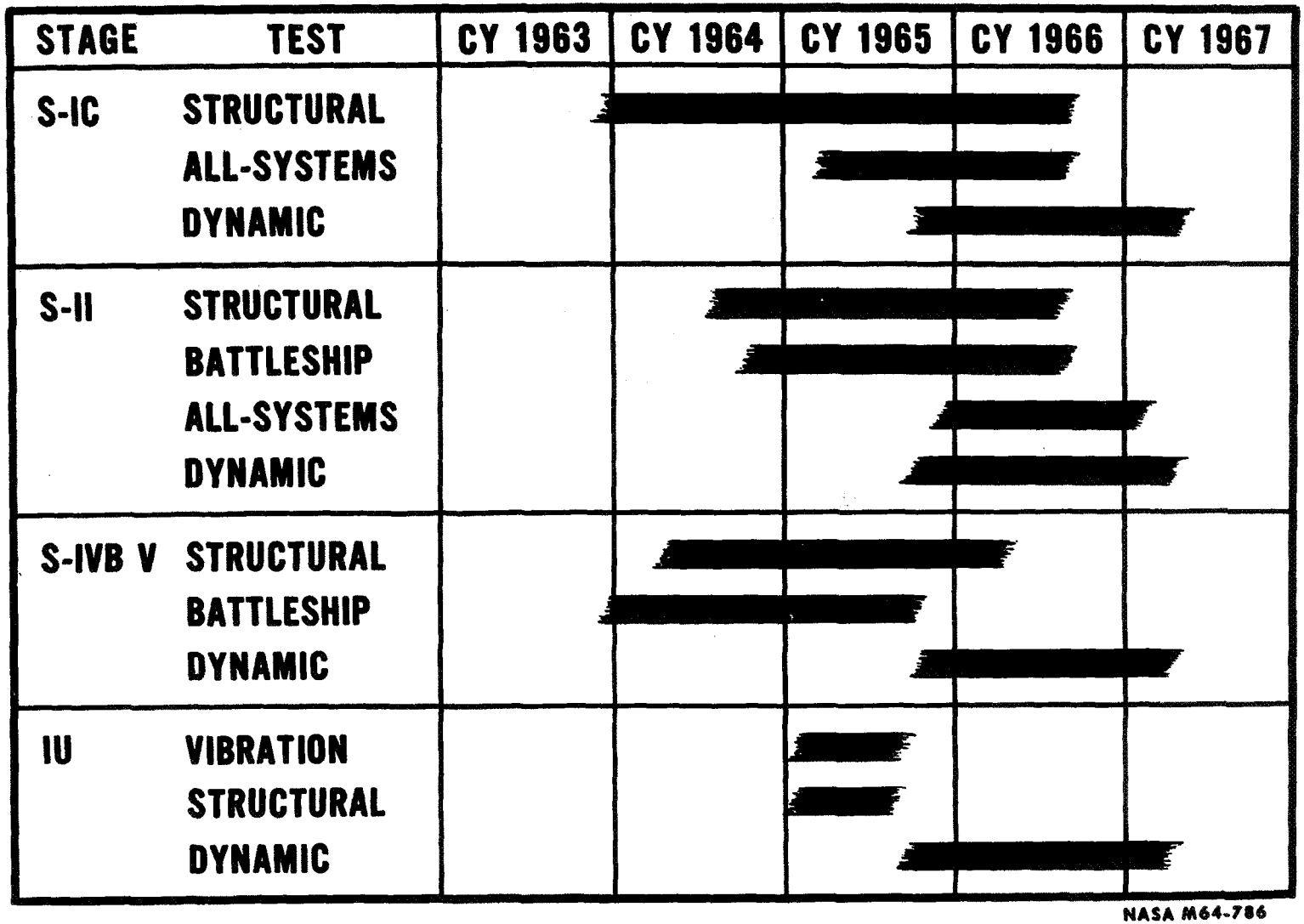

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FIGURE 5

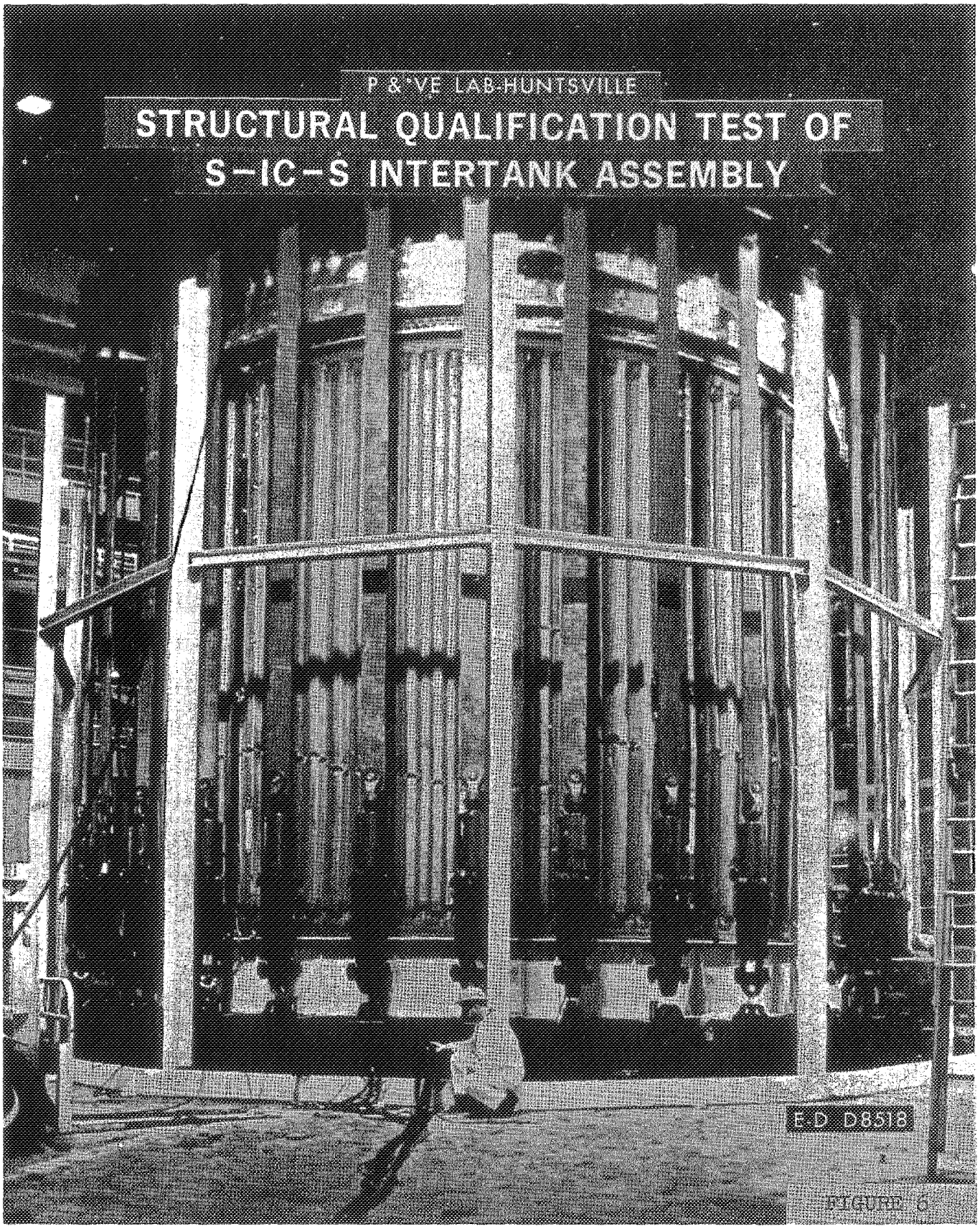

P & VE LAB-HUNTSVILLE
STRUCTURAL QUALIFICATION TEST OF
S-IC-S INTERTANK ASSEMBLY
E D D851B

FIGURE 6

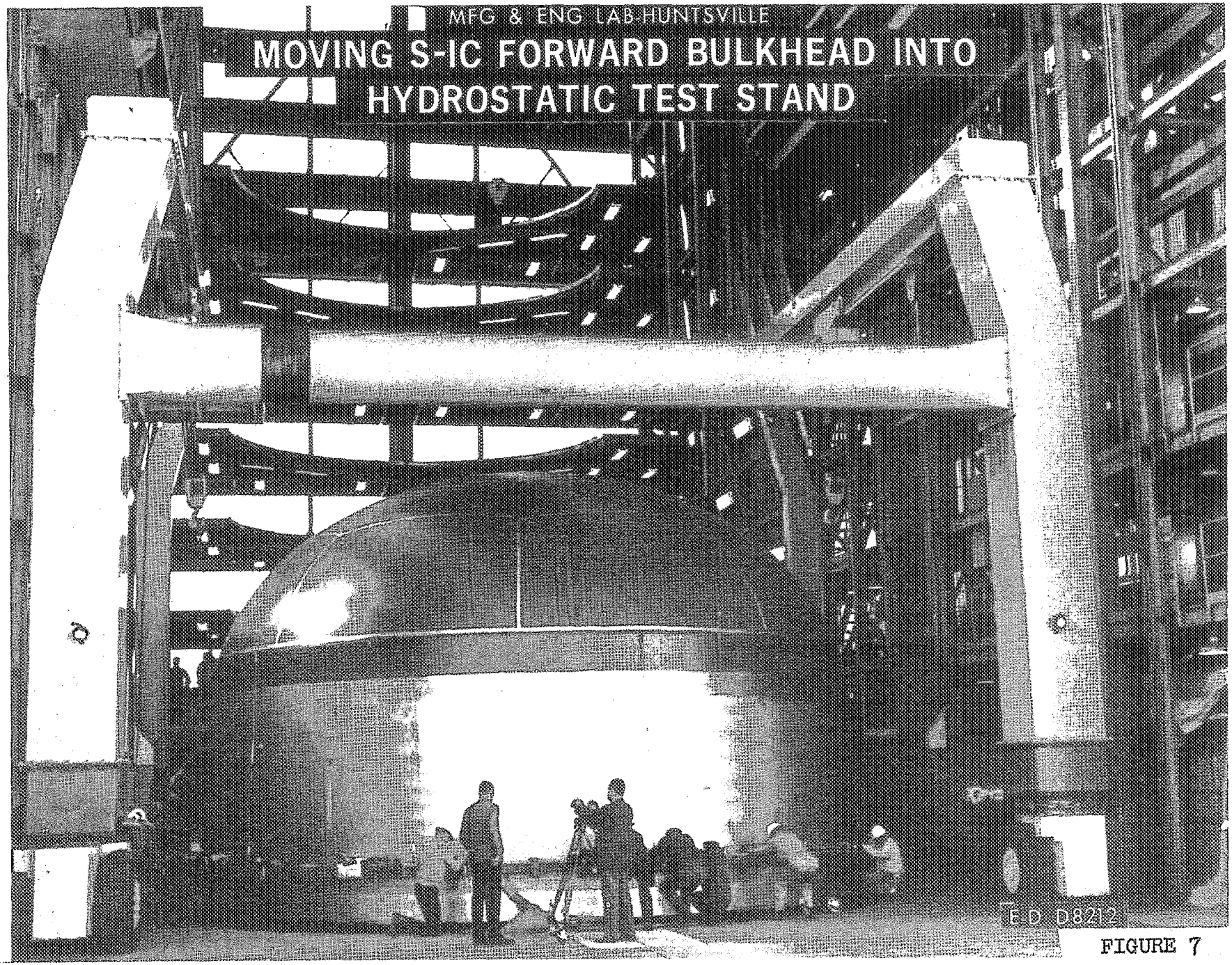

MFG & ENG LAB HUNTSVILLE
MOVING S-IC FORWARD BULKHEAD INTO HYDROSTATIC TEST STAND
E-D D8212

FIGURE 7

S-IC-T BEING INSTALLED IN DEV. TEST STAND HUNTSVILLE

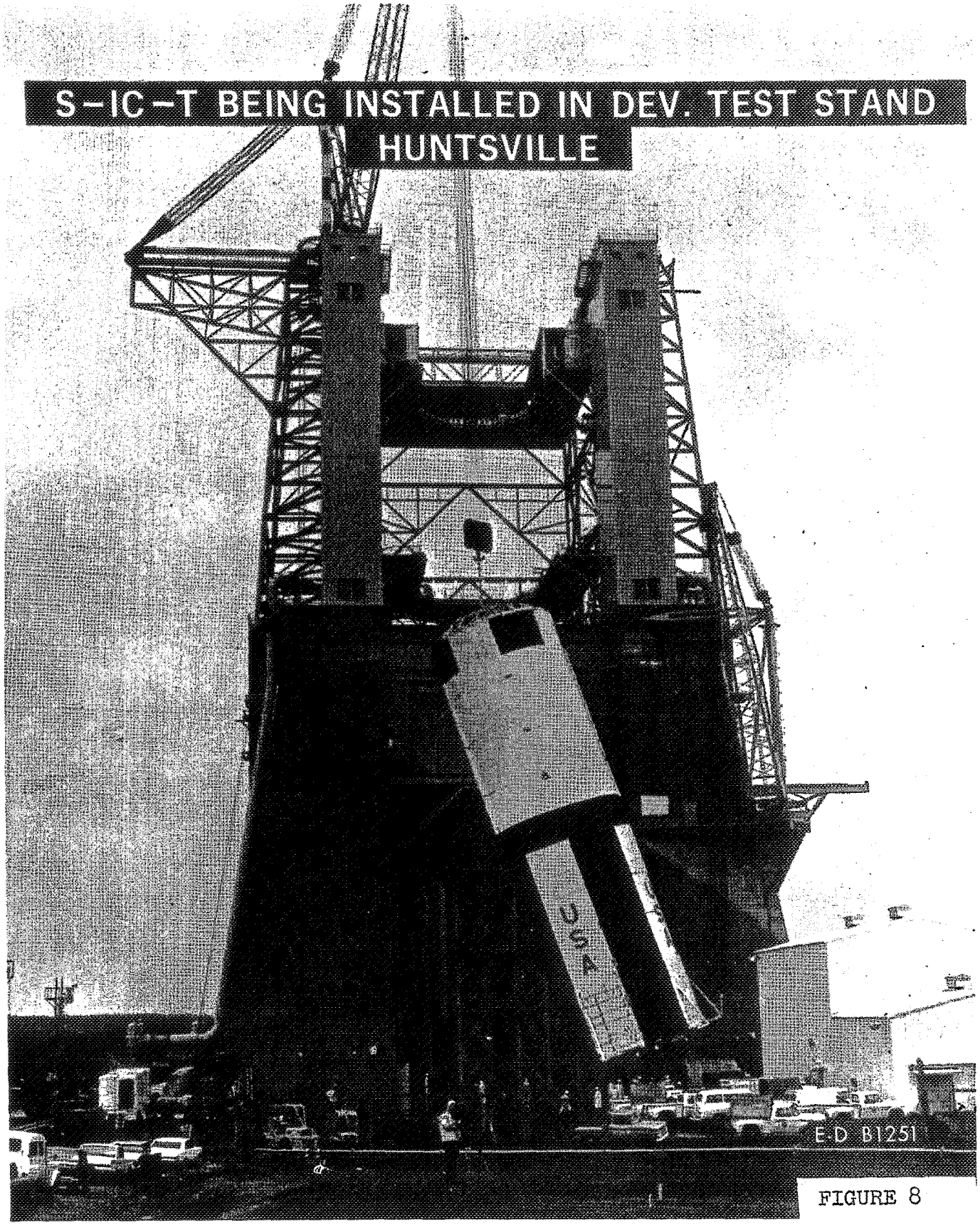


FIGURE 8

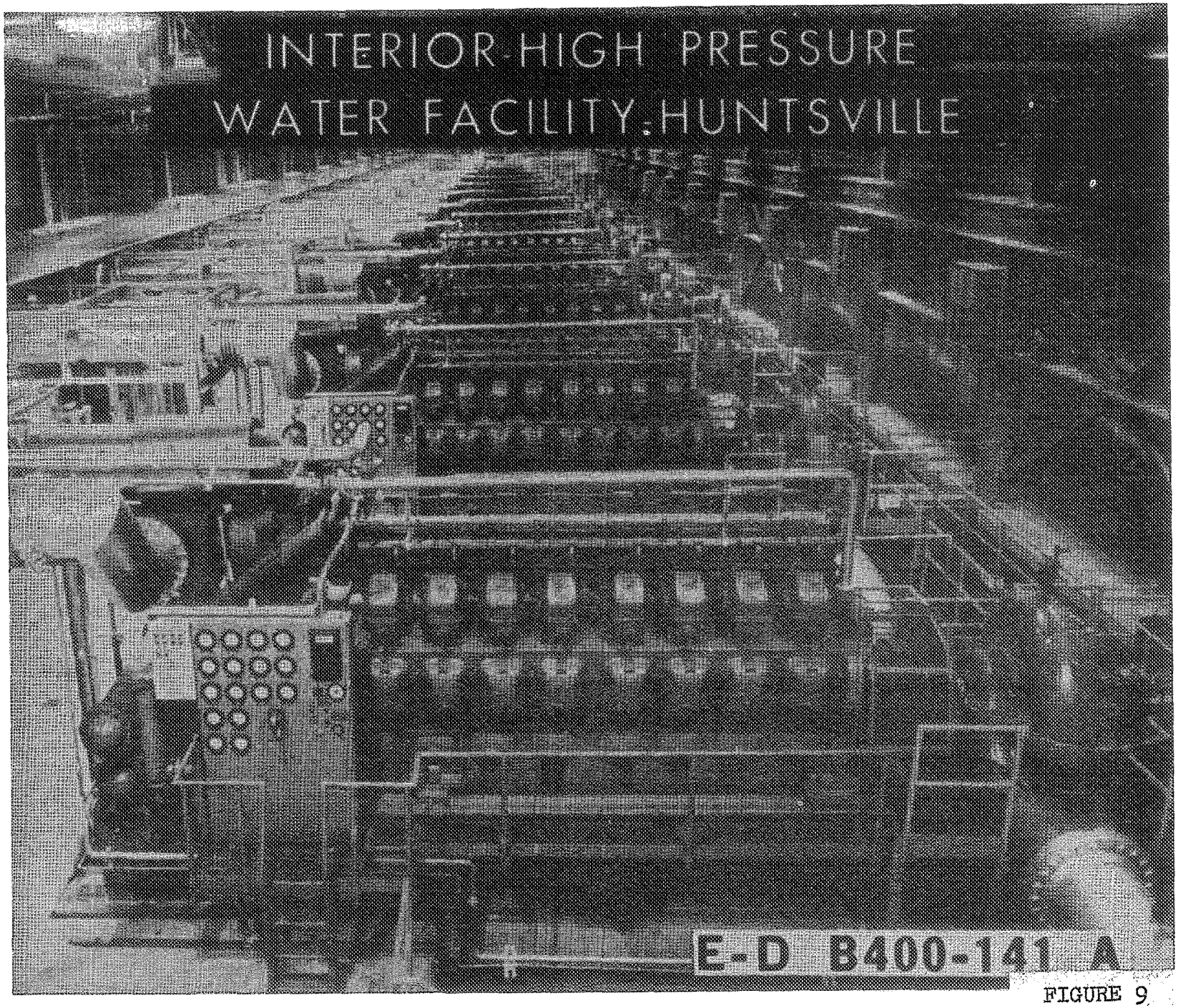
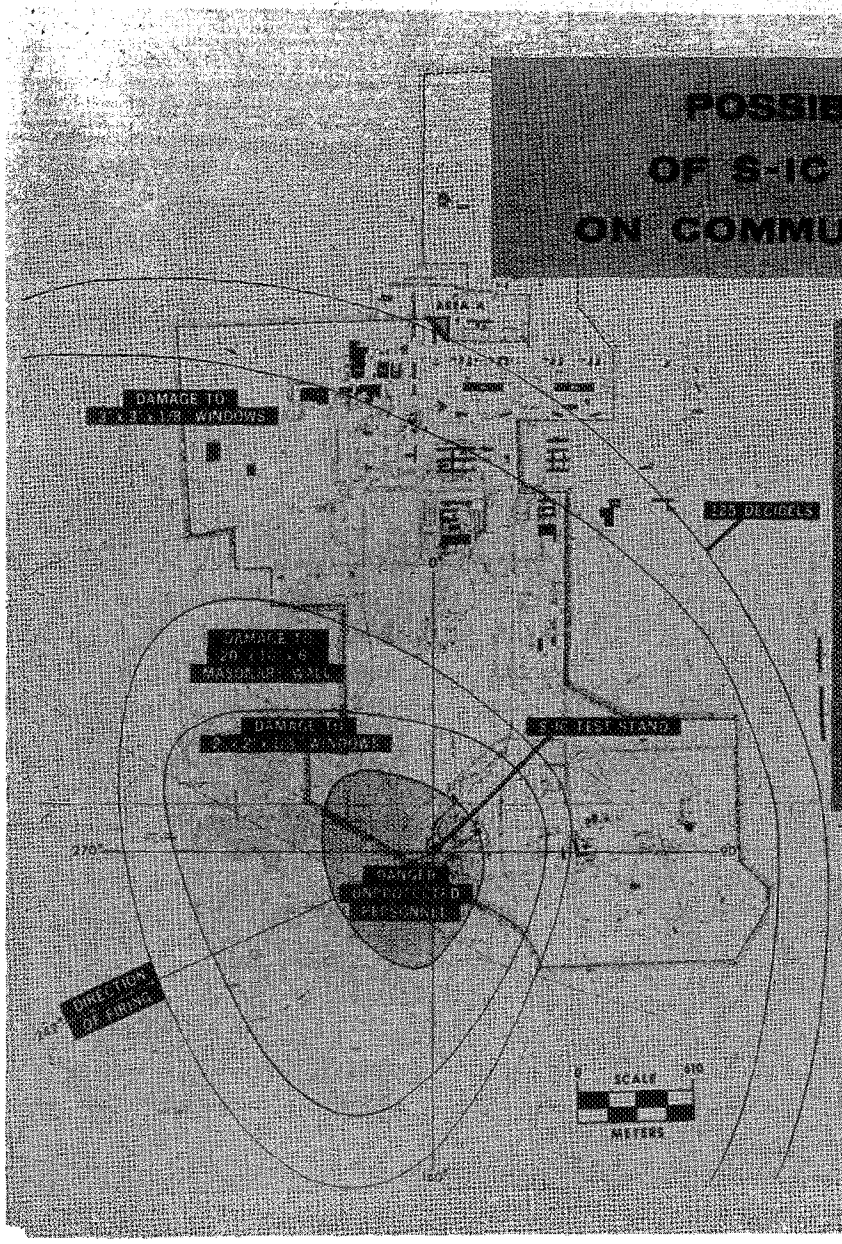
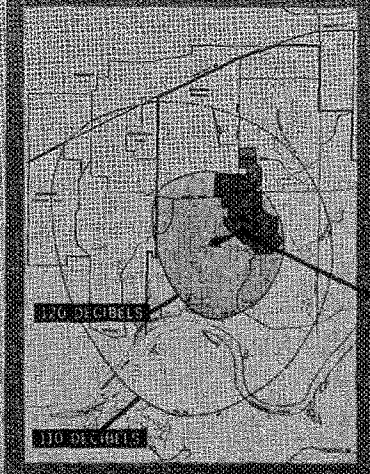

INTERIOR HIGH PRESSURE
WATER FACILITY-HUNTSVILLE
E-D B400-141 A

FIGURE 9

POSSIBLE EFFECTS OF S-IC TEST FIRING ON COMMUNITY PROPERTY





NOTE: WEATHER CONDITIONS HAVE NOT BEEN CONSIDERED IN ABOVE ZONES AND THESE ZONES ARE APPROXIMATE ONLY.

FIGURE 10

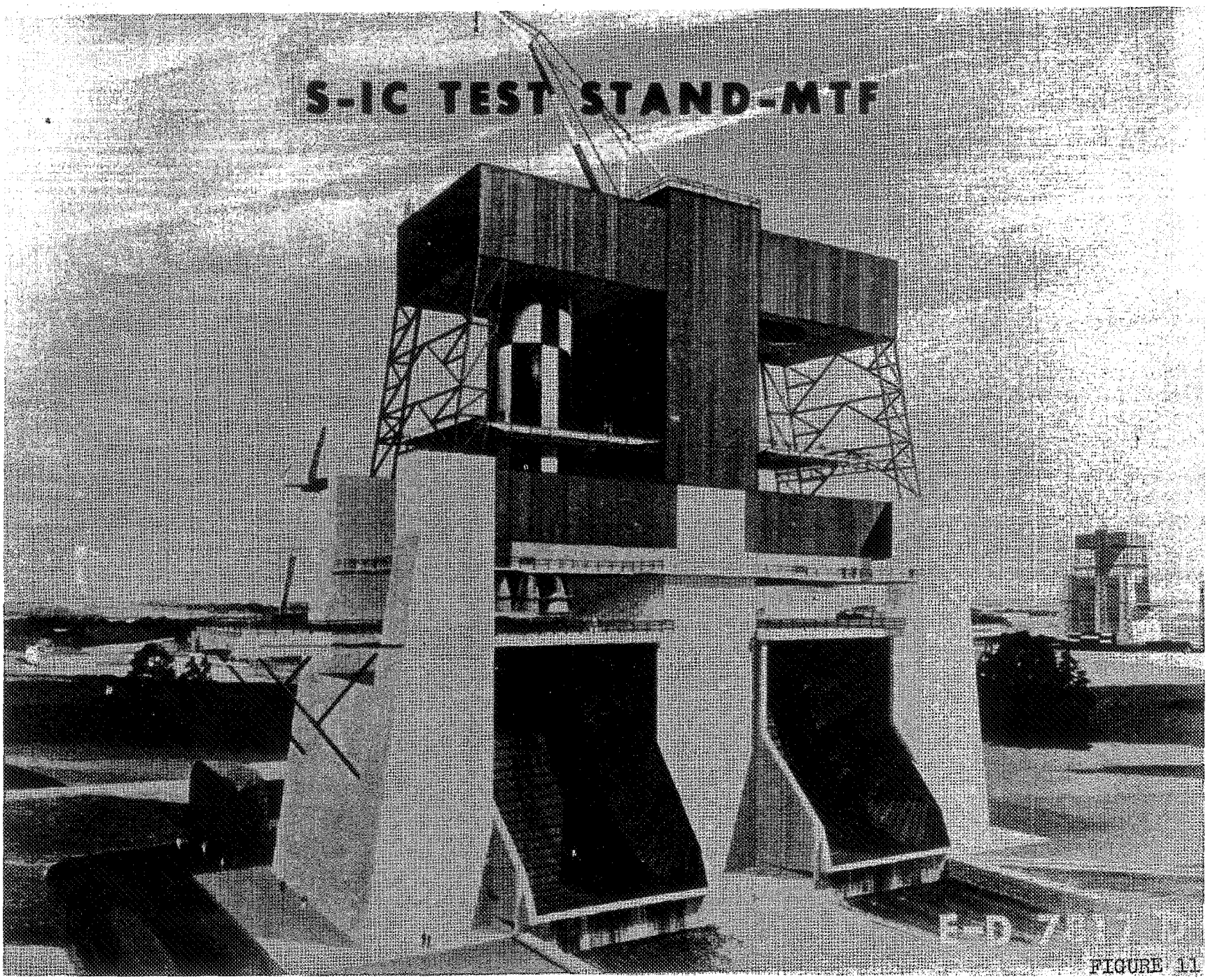


FIGURE 11

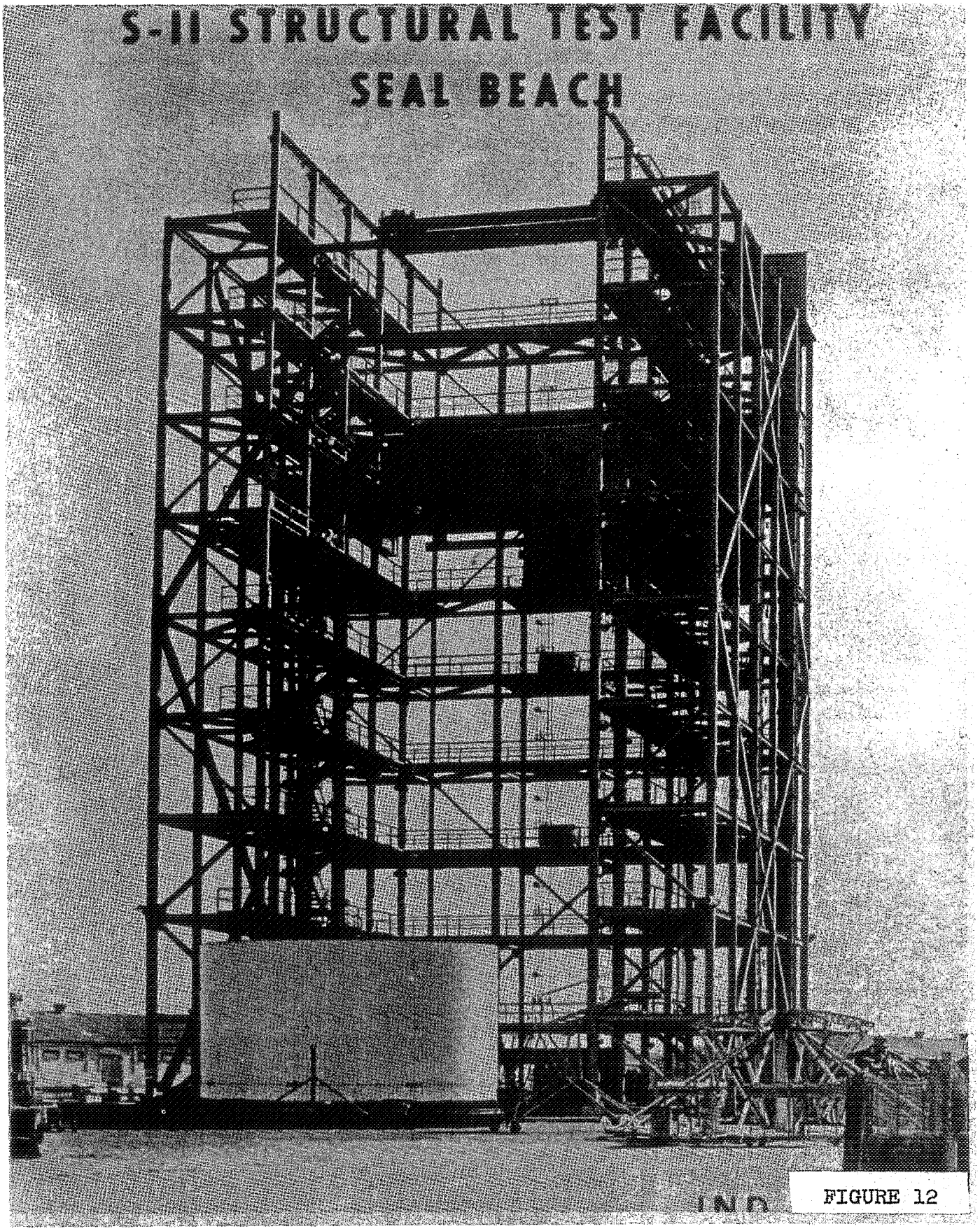


FIGURE 12

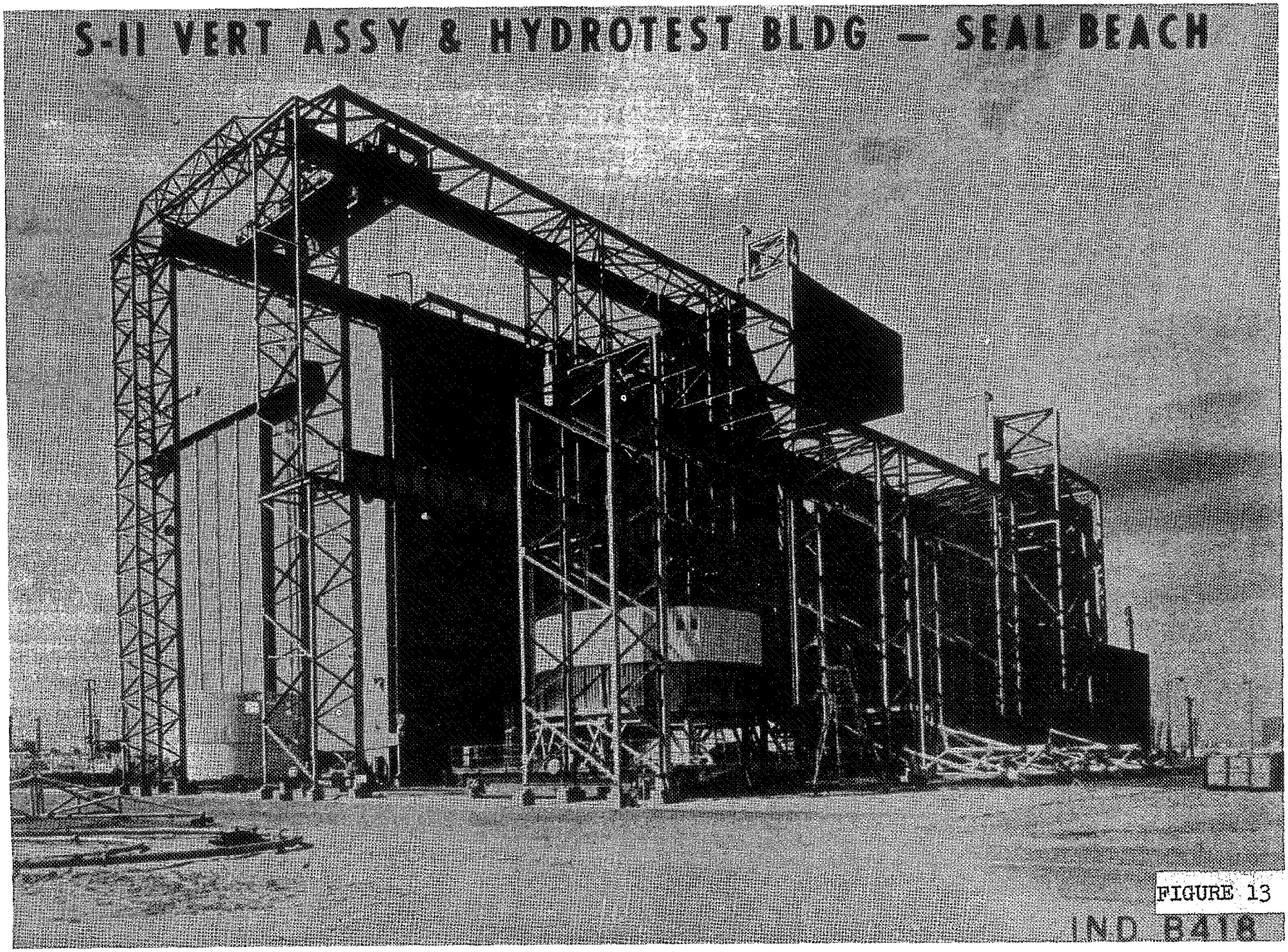
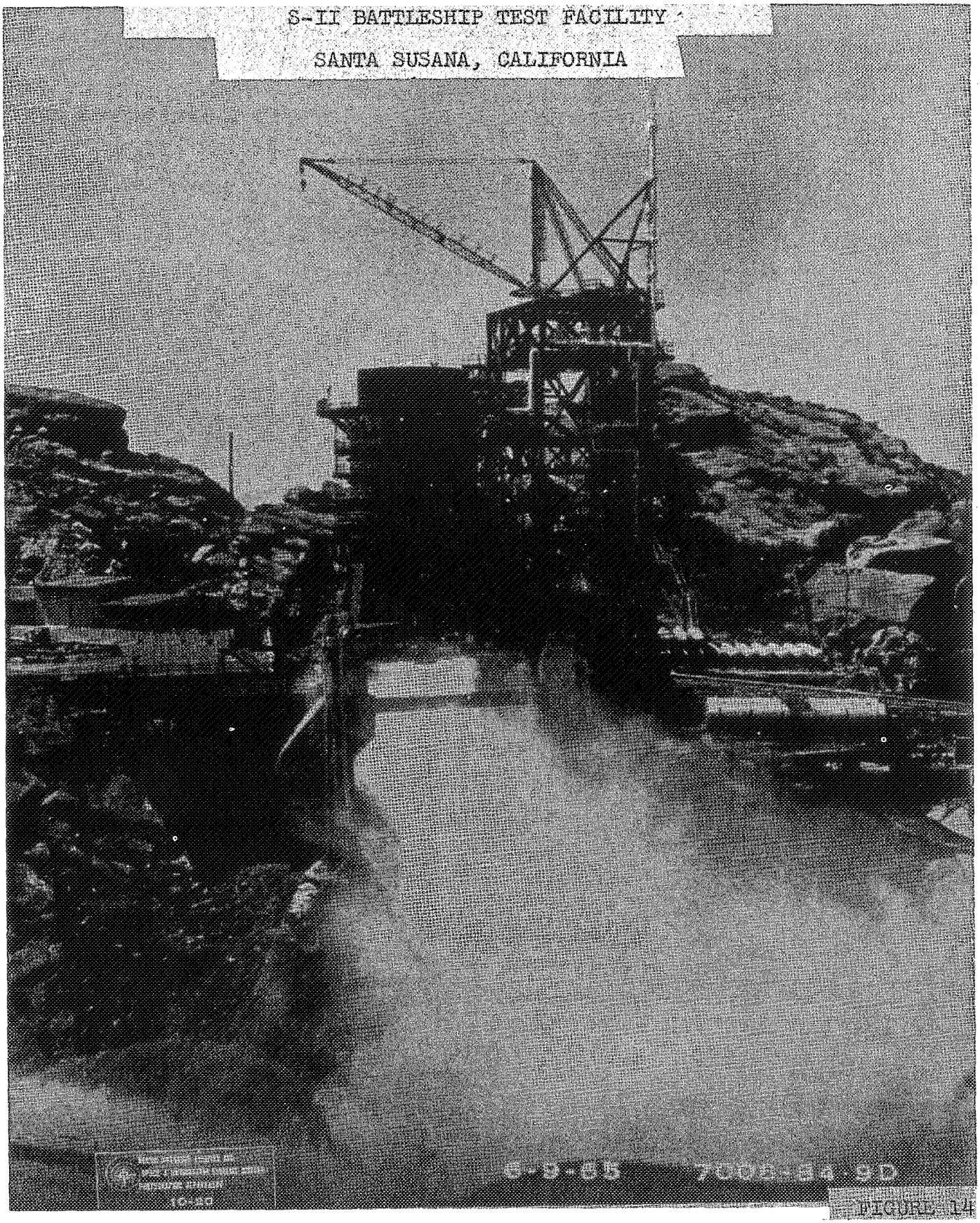


FIGURE 13

IND B418

S-II BATTLESHIP TEST FACILITY

SANTA SUSANA, CALIFORNIA

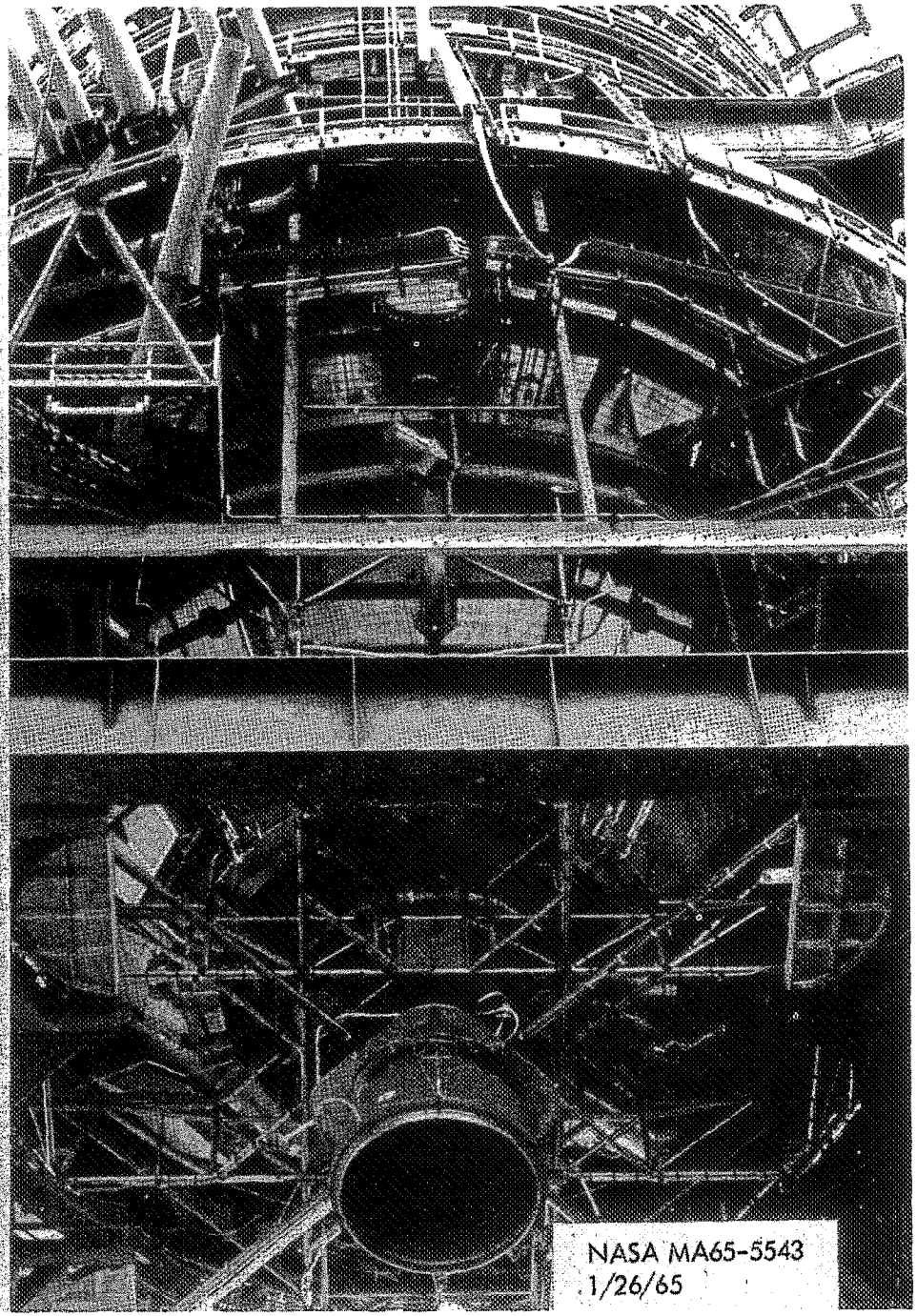


S-II-65 7008-84 SD

FIGURE 14

UNDERSIDE VIEW OF J-2 TEST ENGINE IN BATTLESHIP TEST STAND

SANTA SUSANA, CALIF.
DEC 1964



NASA MA65-5543
1/26/65

FIGURE 15

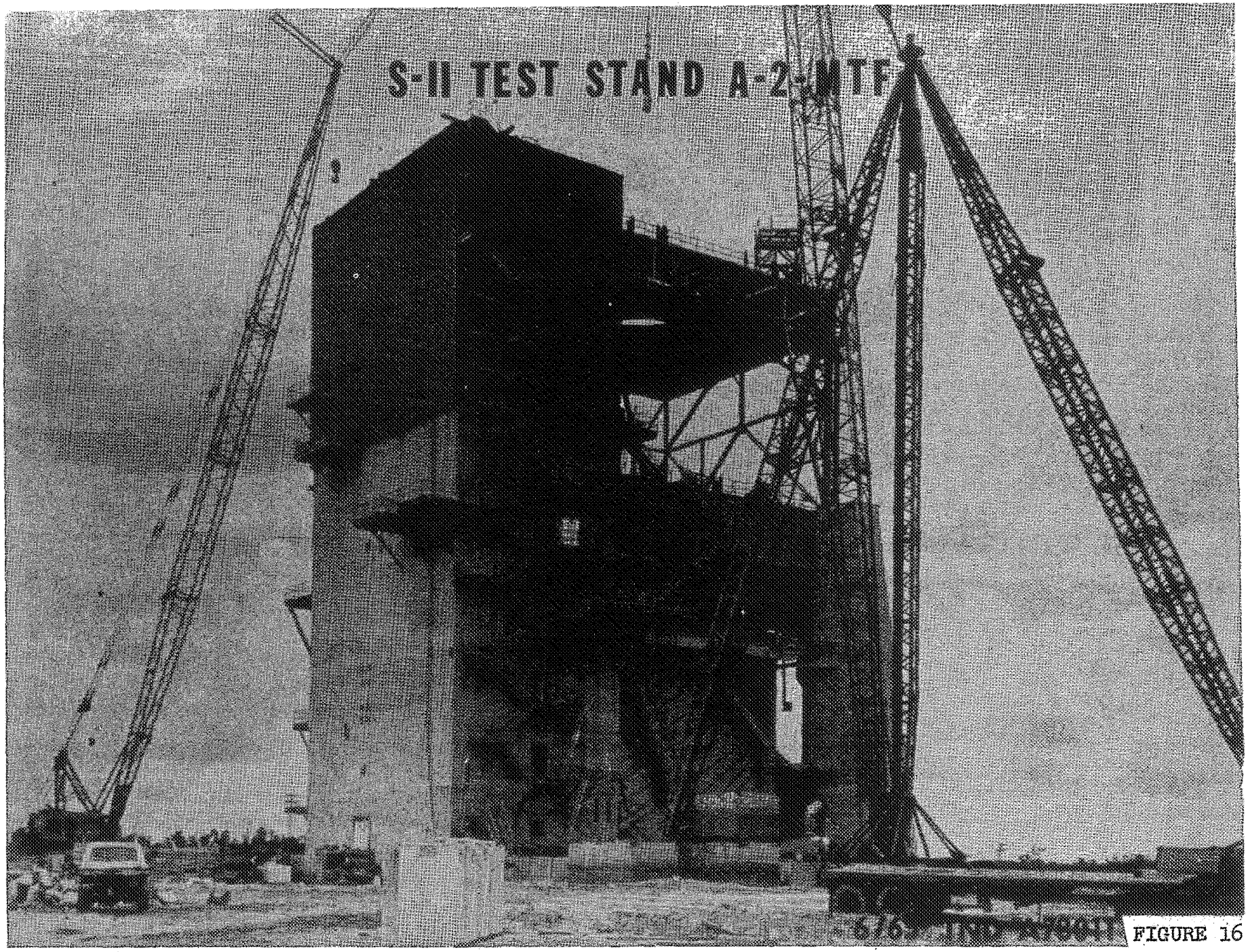


FIGURE 16

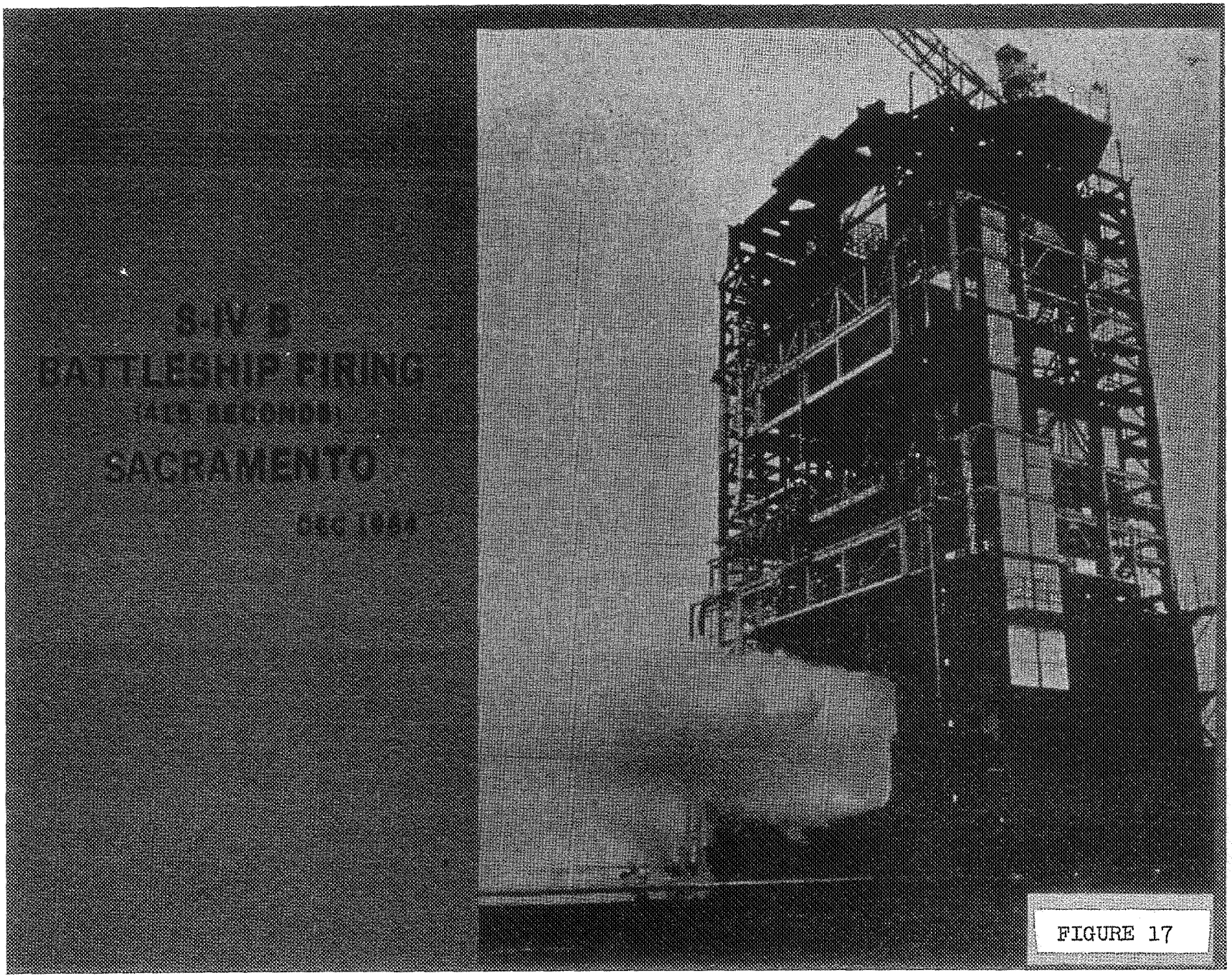
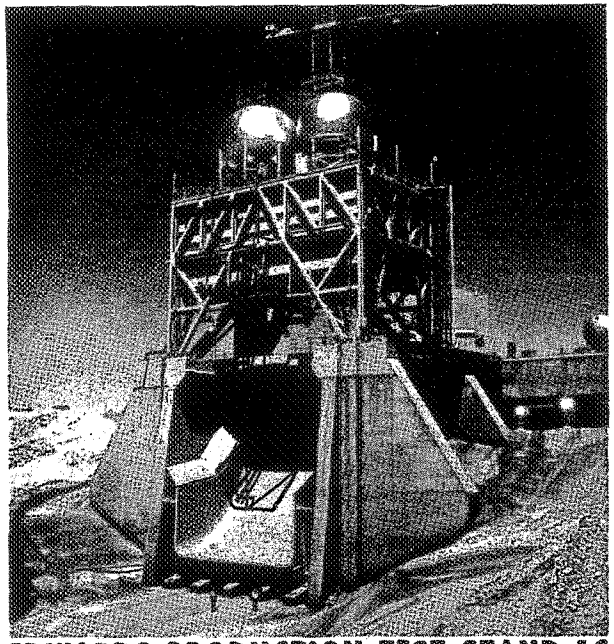
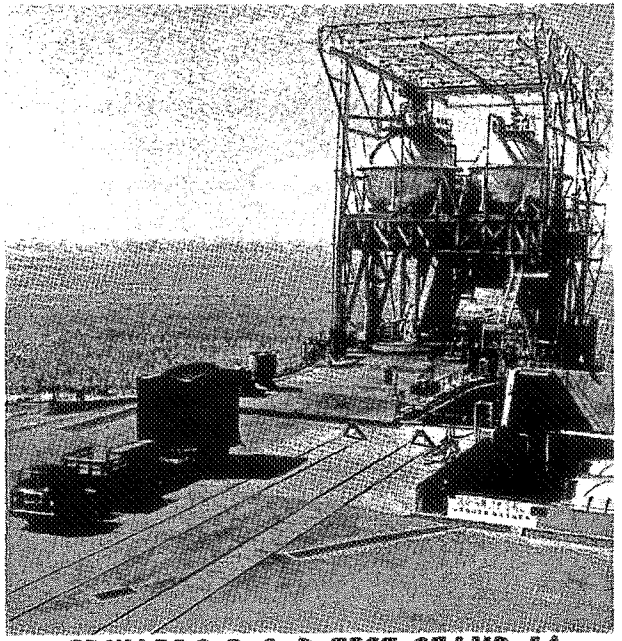


FIGURE 17

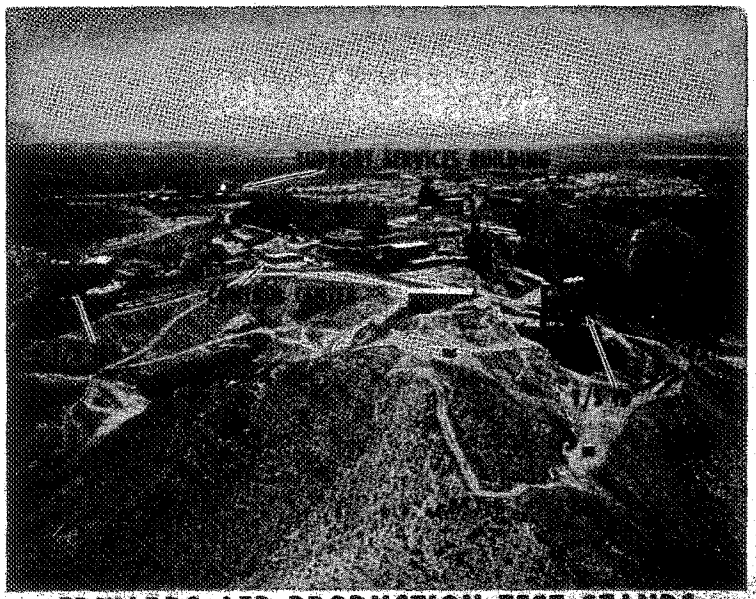
F-1 ENGINE TEST FACILITIES



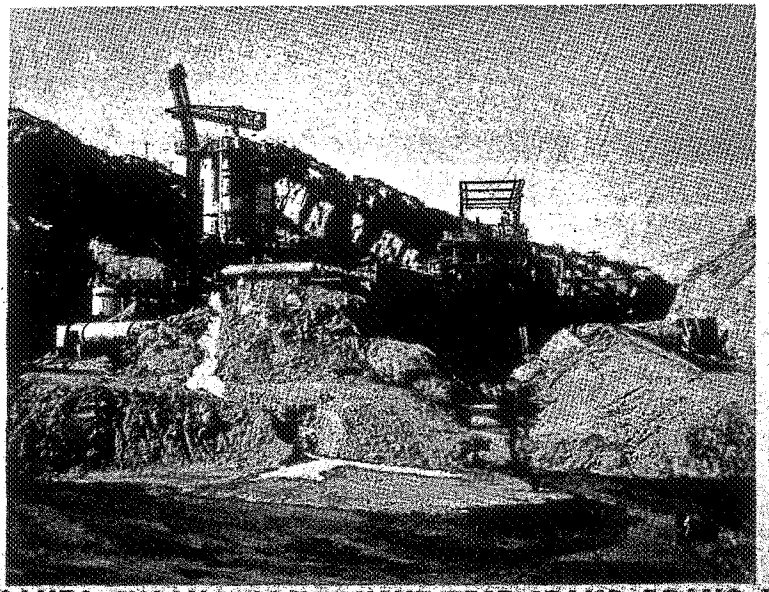
EDWARDS PRODUCTION TEST STAND 1C



EDWARDS R & D TEST STAND 1A



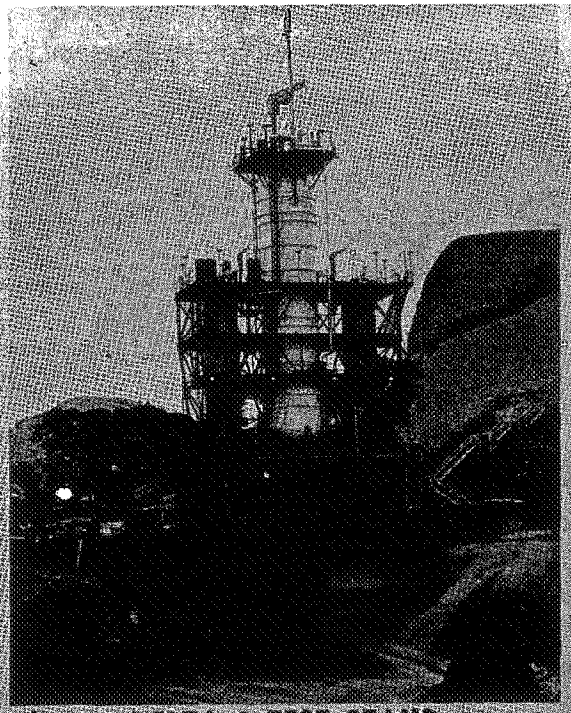
EDWARDS AFB PRODUCTION TEST STANDS



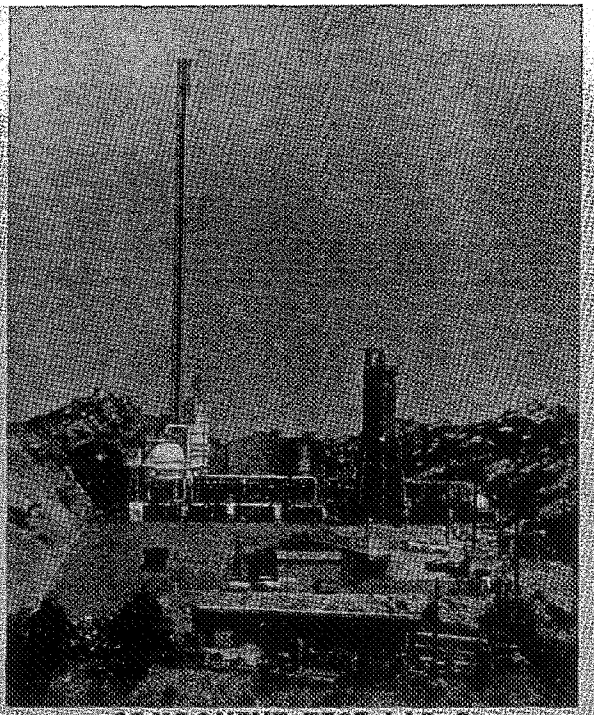
SANTA SUSANA COMPONENT TEST STAND BRAVO 1

FIGURE 18

J-2 TEST FACILITIES

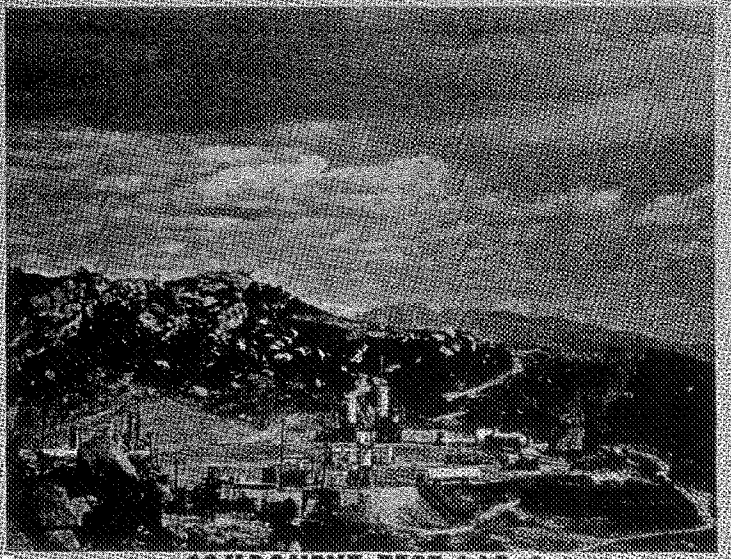


DELTA 2 TEST STAND



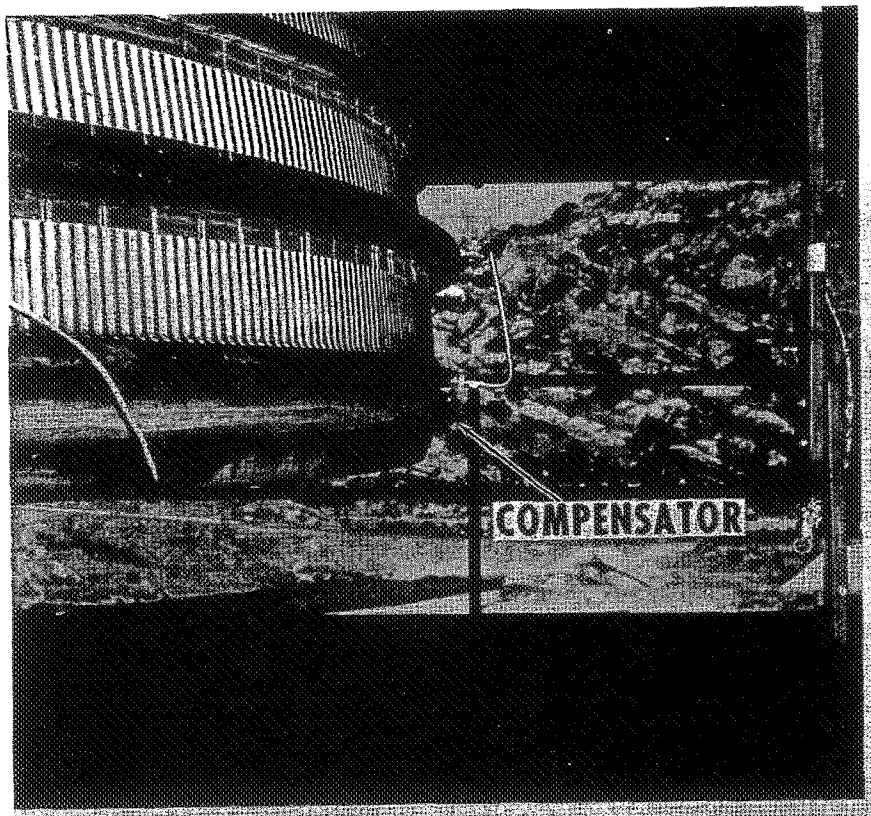
COMPONENT TEST LAB 5

BOWL AREA TEST STANDS



COMPONENT TEST LAB 3

FIGURE 19



J-2 ENGINE MOUNTED COMPENSATOR

J-2 ENGINE FIRING WITH HOLD DOWN ARMS

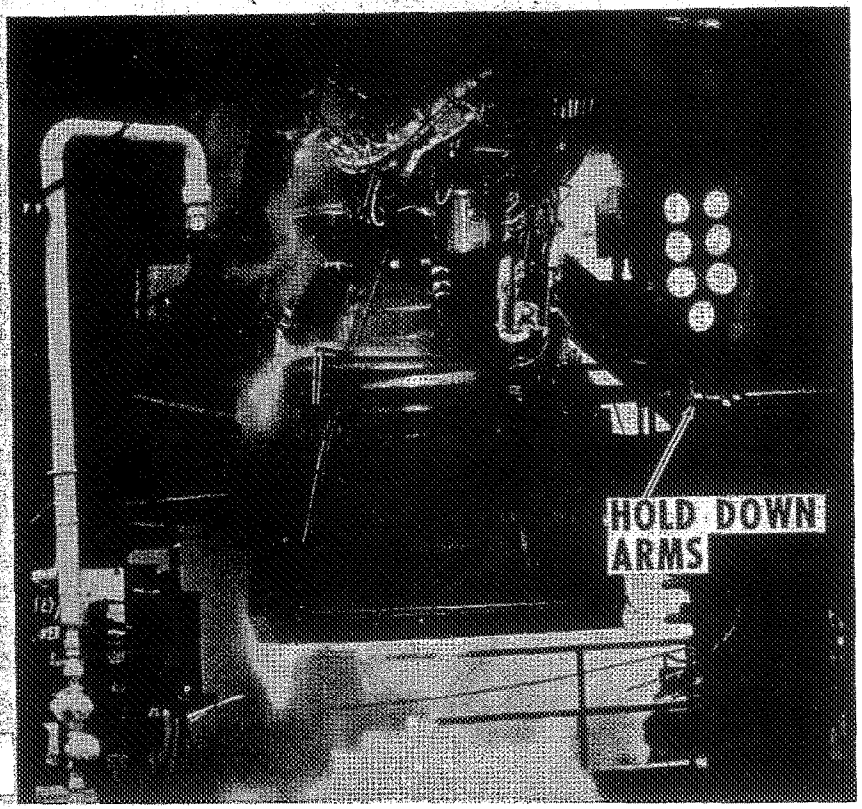


FIGURE 20

J-2 ENGINE ALTITUDE TEST FACILITY

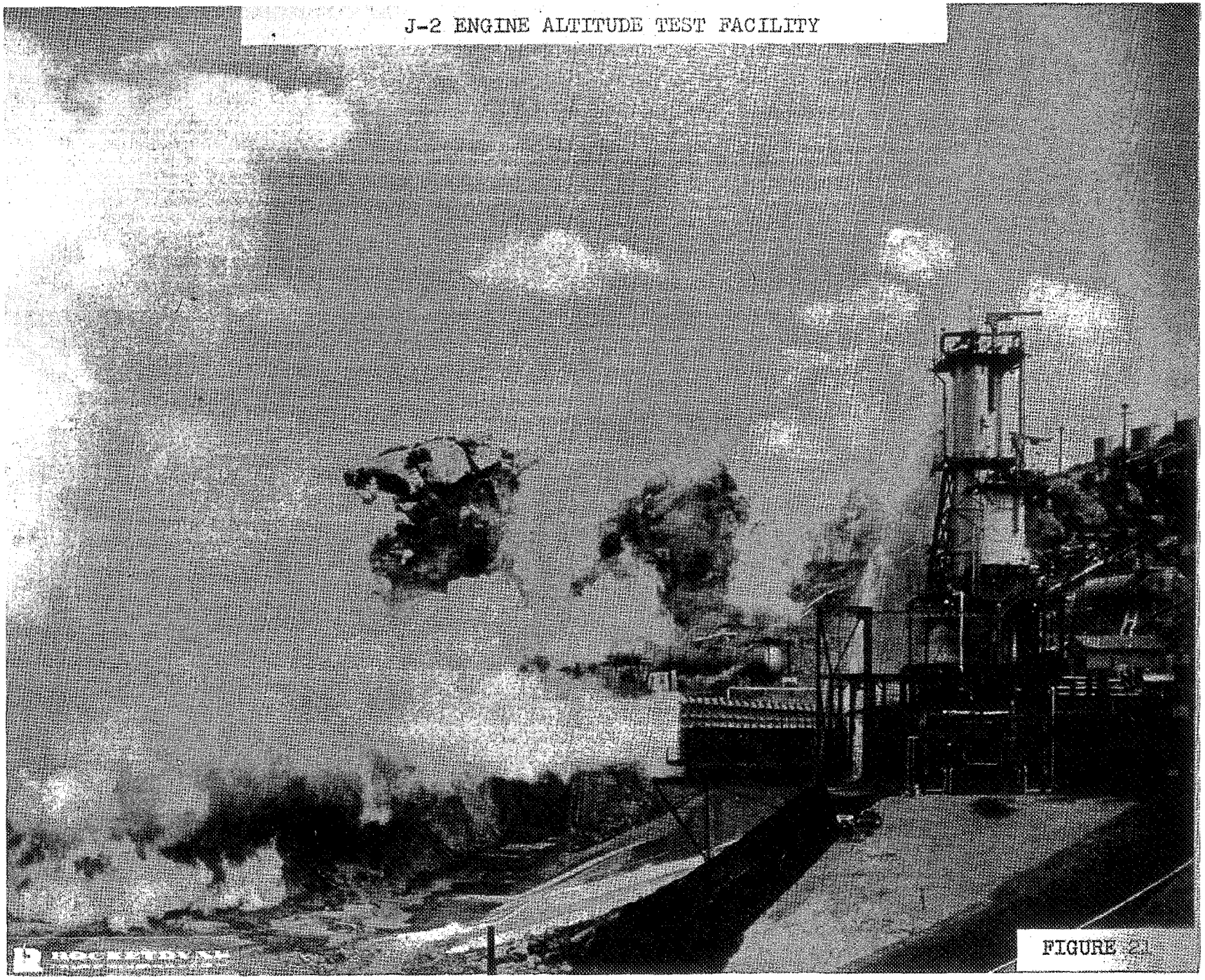


FIGURE 21

DATA FLOW CHART FOR DESIGN OF SATURN V VEHICLE CONTROL SYSTEM

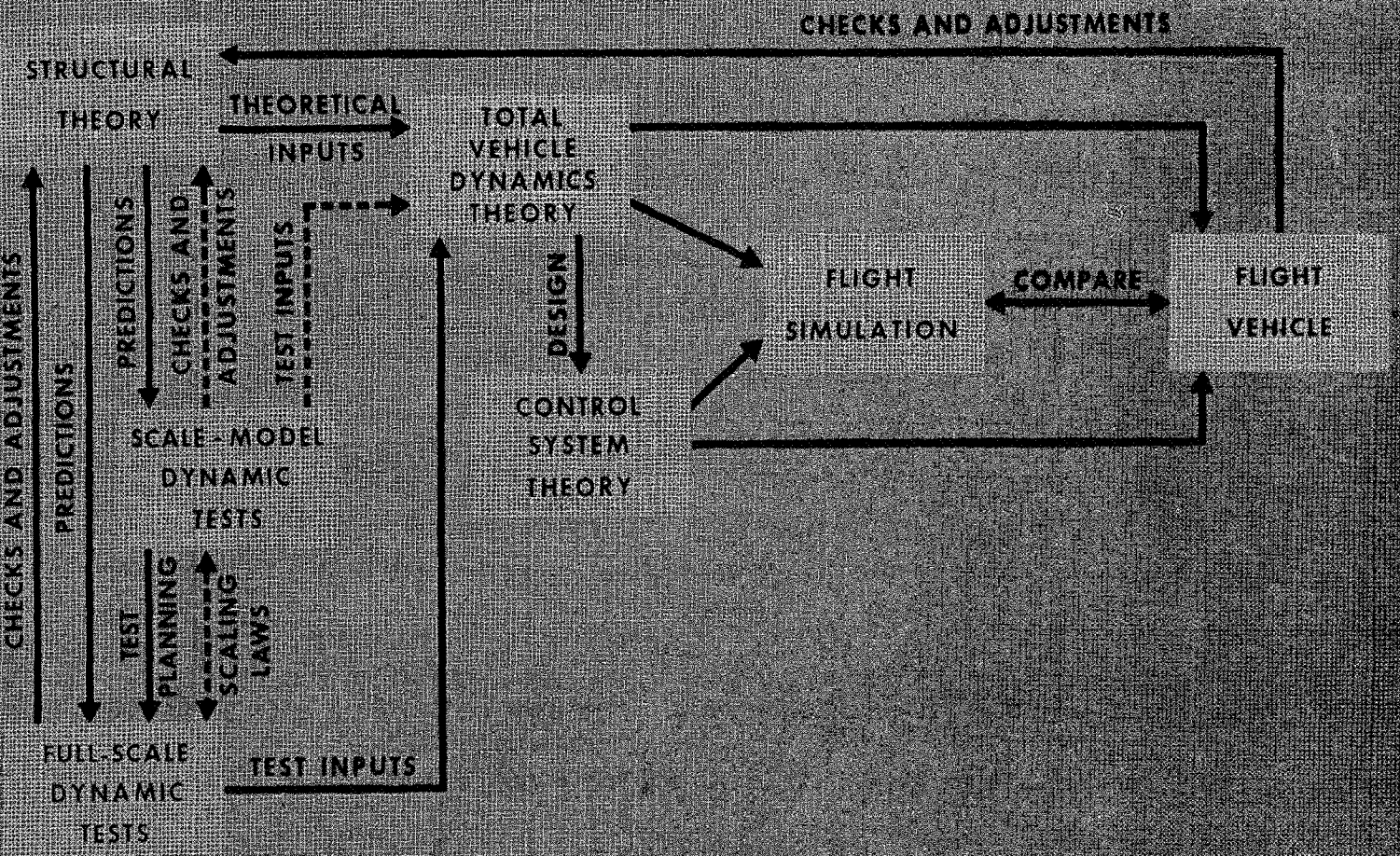


FIGURE 22

SATURN I VEHICLE IN DYNAMIC TEST STAND

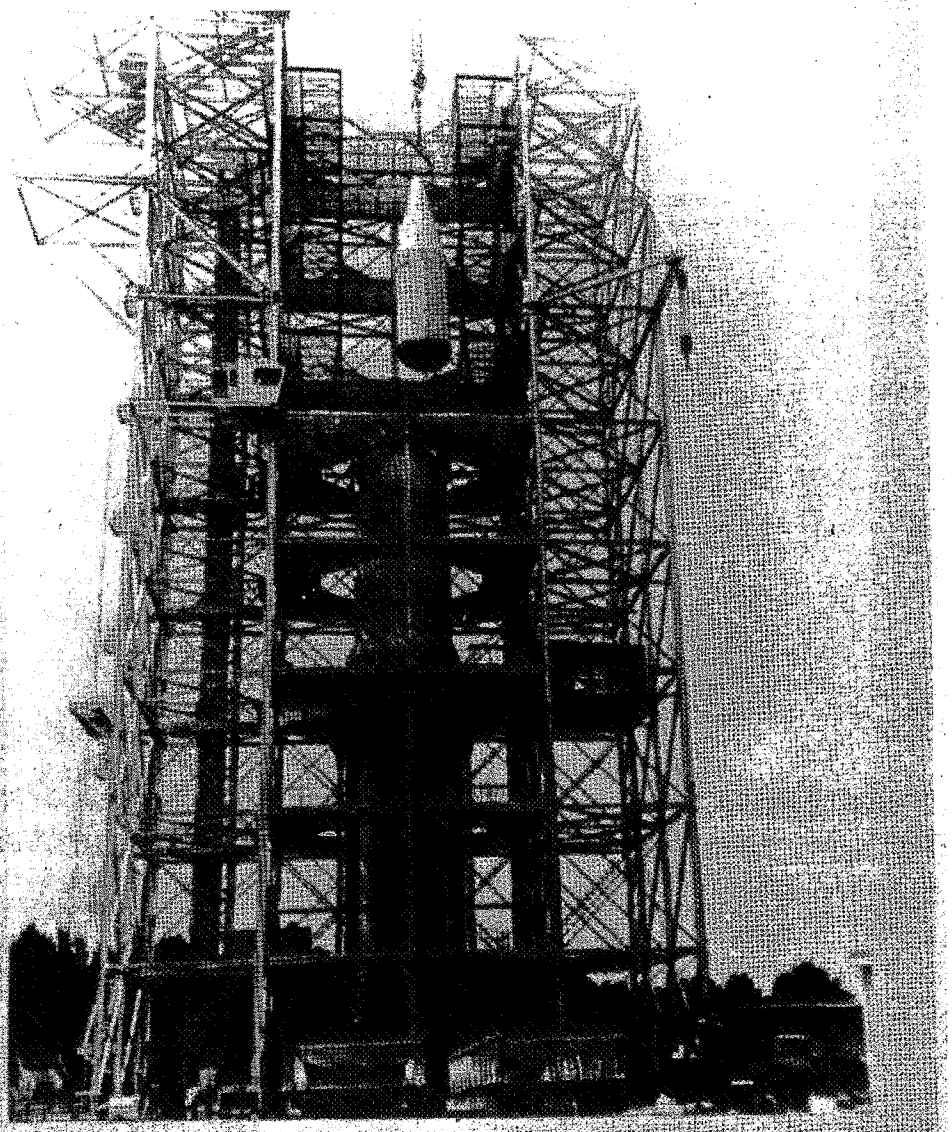


FIGURE 23

DETAILS OF SATURN I VEHICLE IN DYNAMIC TEST STAND

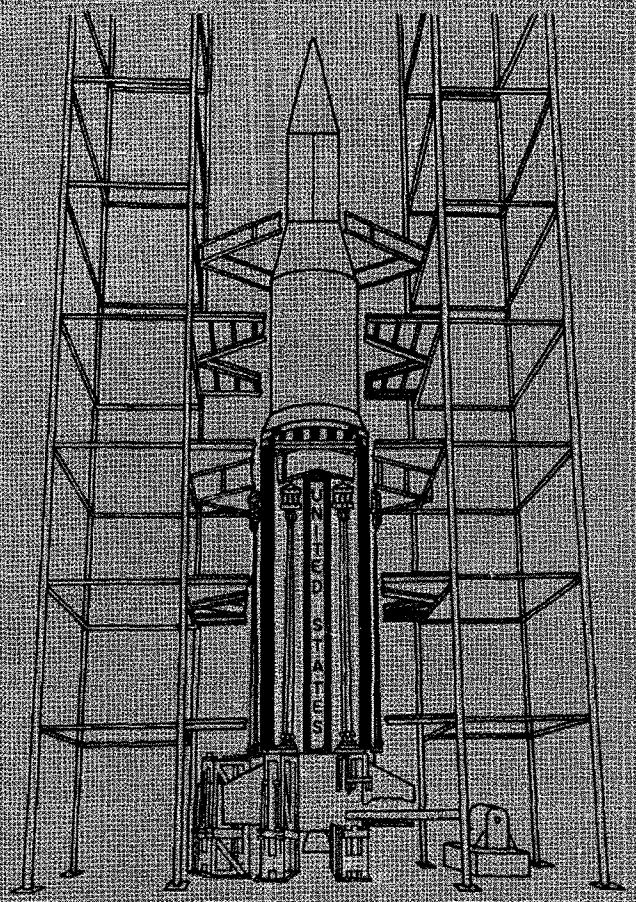


FIGURE 24

SATURN DYNAMIC TEST STANDS–HUNTSVILLE
SAT V
SAT I/IB
ED B432 L

FIGURE 25

IMPROVED SUPPORT SYSTEM FOR SATURN V DYNAMIC TEST

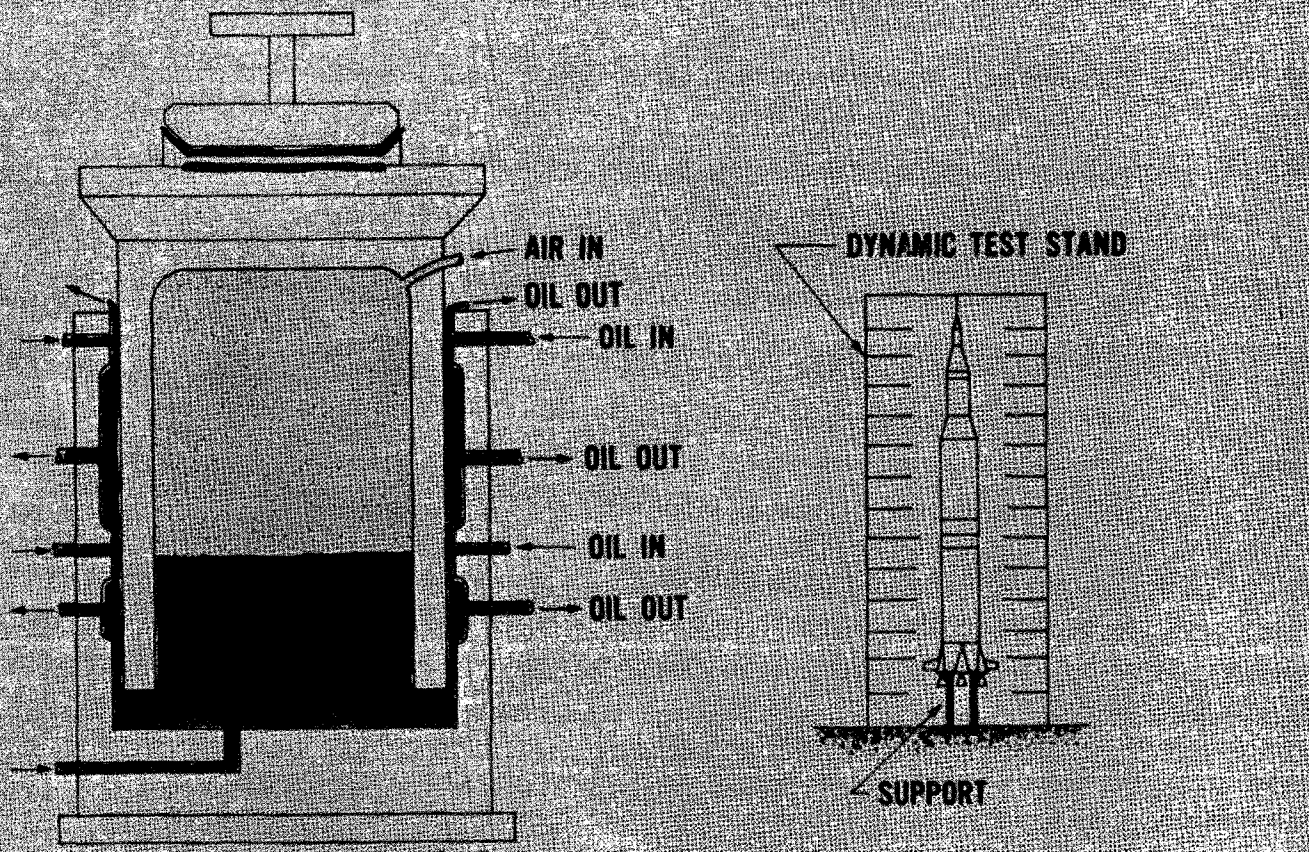


FIGURE 26

SCALE MODEL OF SATURN V VEHICLE

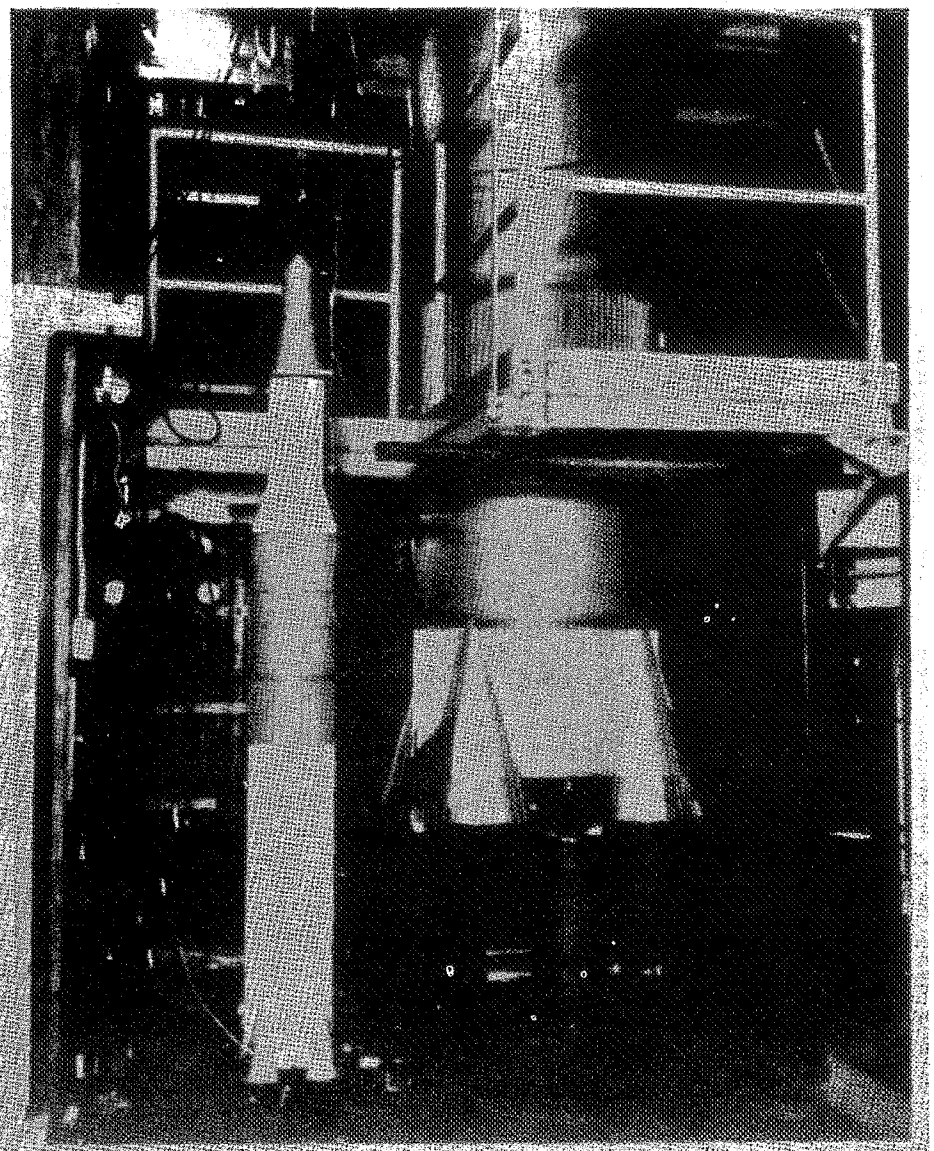


FIGURE 27

SPACECRAFT DEVELOPMENT GROUND TEST REQUIREMENTS

COMMAND MODULE
- LAND AND WATER IMPACT
- LAUNCH AND REENTRY HEATING
- LAUNCH AND REENTRY LOADS
- VIBRATION AND ACOUSTIC
- AERODYNAMIC CHARACTERISTICS
- RENDEZVOUS AND DOCKING
- THERMAL VACUUM

SERVICE MODULE
- LAUNCH HEATING AND LOADS
- VIBRATION AND ACOUSTIC
- THERMAL VACUUM
- PROPULSION SYSTEM PERFORMANCE

LUNAR EXCURSION MODULE
- EARTH LAUNCH LOADS
- LUNAR LANDING AND ASCENT LOADS
- LUNAR LANDING DYNAMICS
- RENDEZVOUS AND DOCKING
- VIBRATION AND ACOUSTIC
- THERMAL VACUUM
- PROPULSION SYSTEM PERFORMANCE

INTEGRATED SPACECRAFT
- EARTH LAUNCH HEATING AND LOADS
- VIBRATION AND ACOUSTIC
- THERMAL VACUUM
- SPACECRAFT DYNAMICS

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FIGURE 28

COMMAND MODULE BOILERPLATE GROUND TEST

WATER IMPACT TEST
NAA DOWNEY - LATE 1963

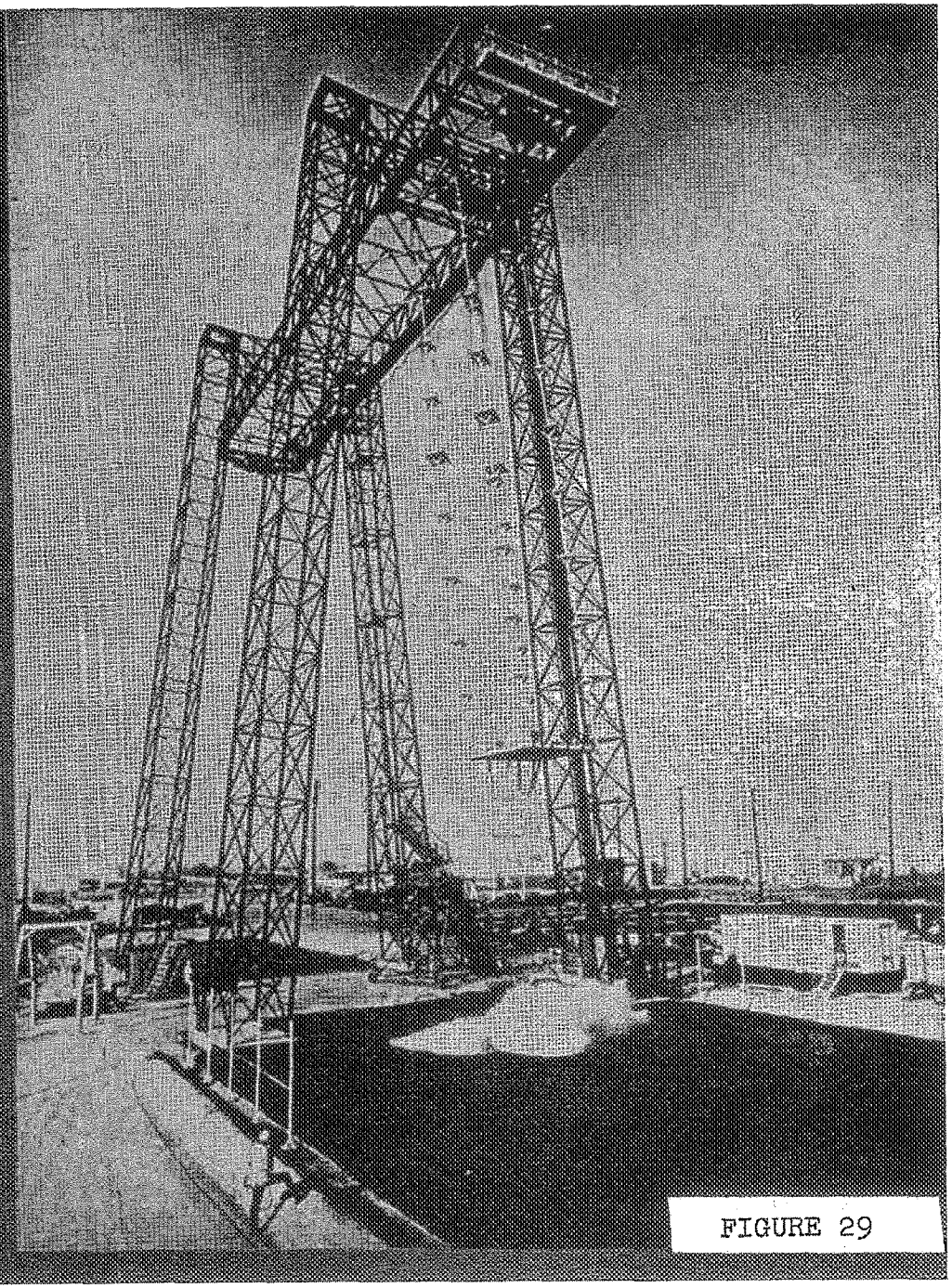


FIGURE 29

APOLLO COMMAND MODULE

THERMAL TEST INSTALLATION

NASA M65-9768

FIGURE 30

FLIGHT ACCELERATION FACILITY

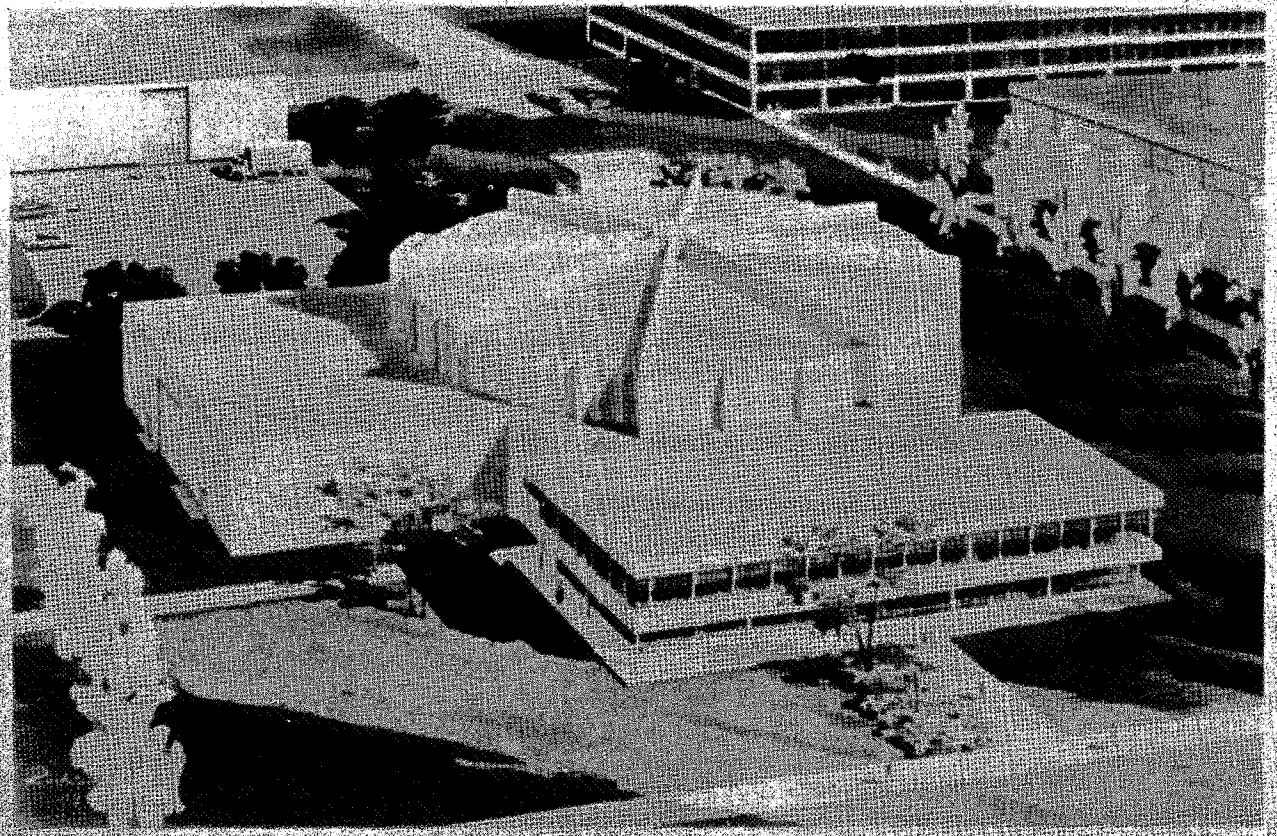


FIGURE 31

CENTRIFUGE SYSTEM

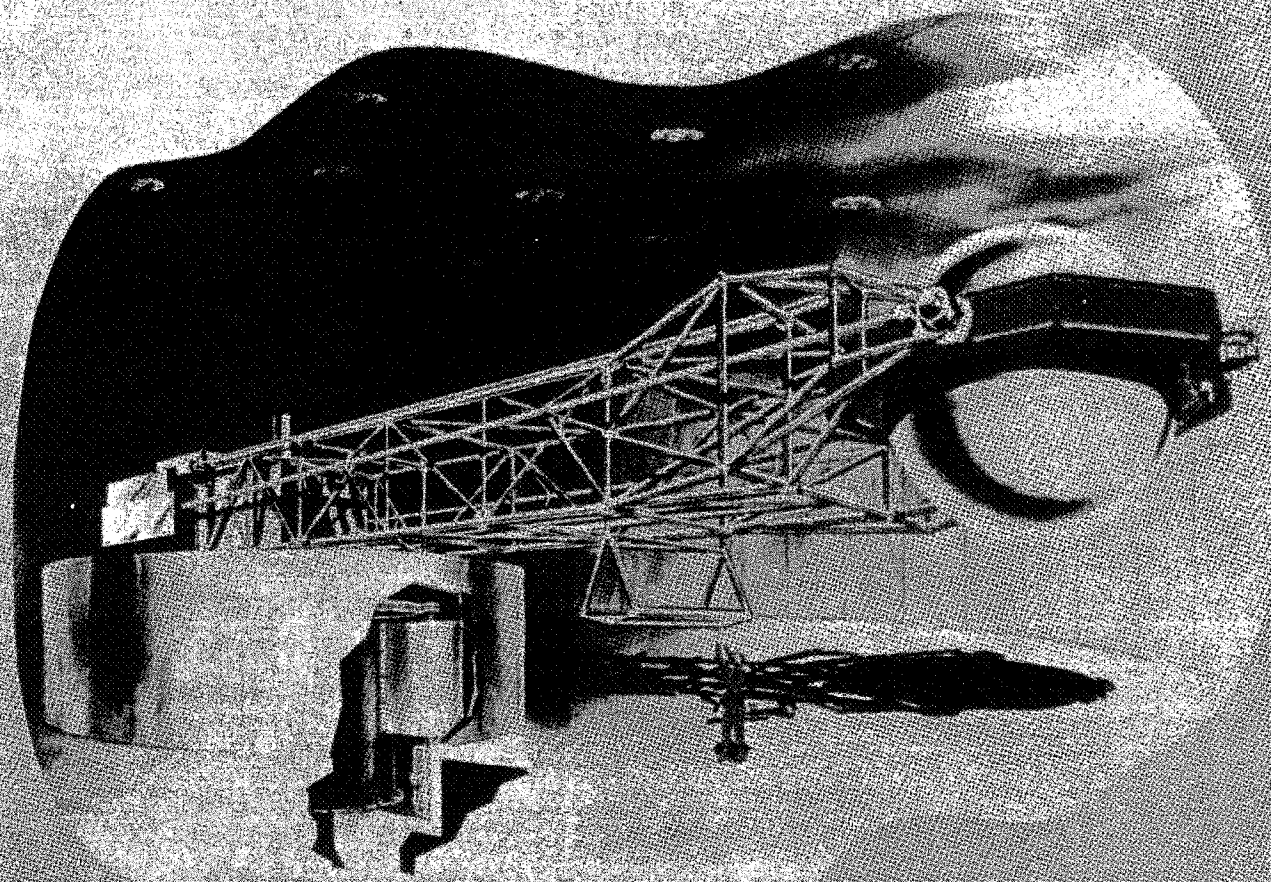


FIGURE 32

SPACE ENVIRONMENT SIMULATION LABORATORY

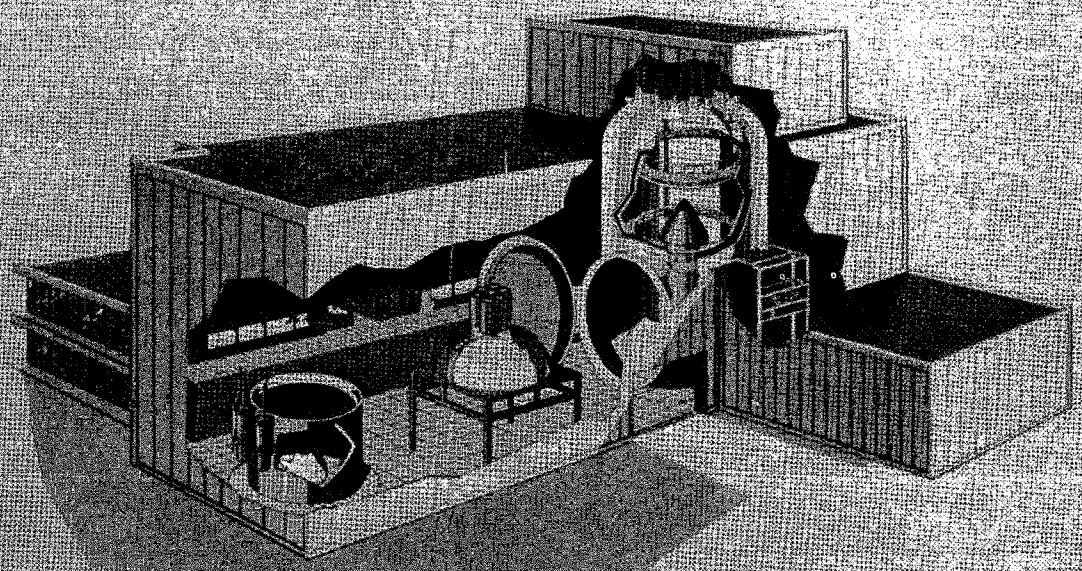


FIGURE 33

SPACE ENVIRONMENT SIMULATION LABORATORY
CHAMBER A

CAPABILITY & DESCRIPTION	
OUTSIDE DIMENSIONS	65 FT DIA×120 FT HIGH
INSIDE CLEAR DIMENSIONS	55 FT DIA×90 FT HIGH
MAXIMUM VEHICLE SIZE	25 FT DIA×65 FT HIGH
MAXIMUM VEHICLE WEIGHT	150,000 POUNDS
PRESSURE LEVEL	1×10^{-5} TORR (80 MILES ALTITUDE)
SOLAR SIMULATION SOURCE	CARBON ARC UNITS
TEMPERATURE INTERIOR CHAMBER WALLS	-280 F
LUNAR PLANE	ROTATES ± 180

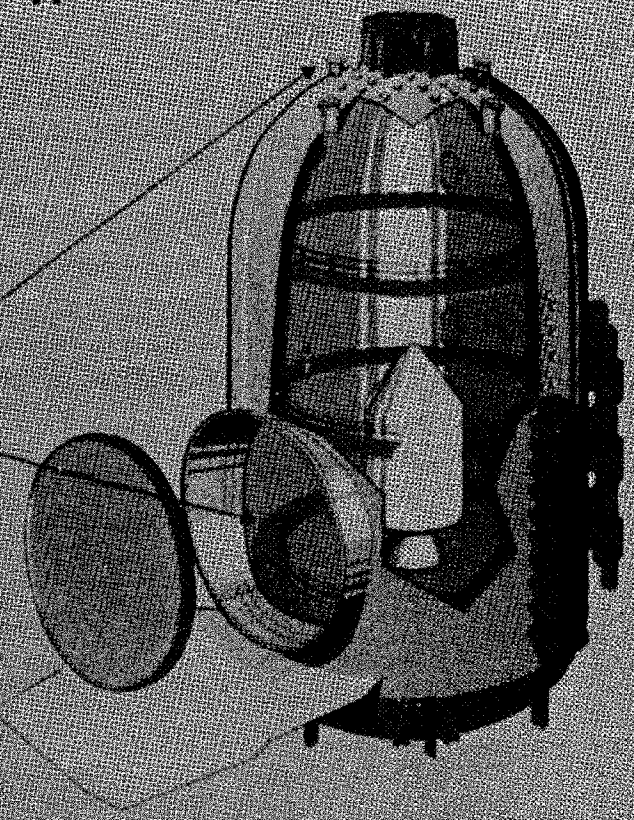
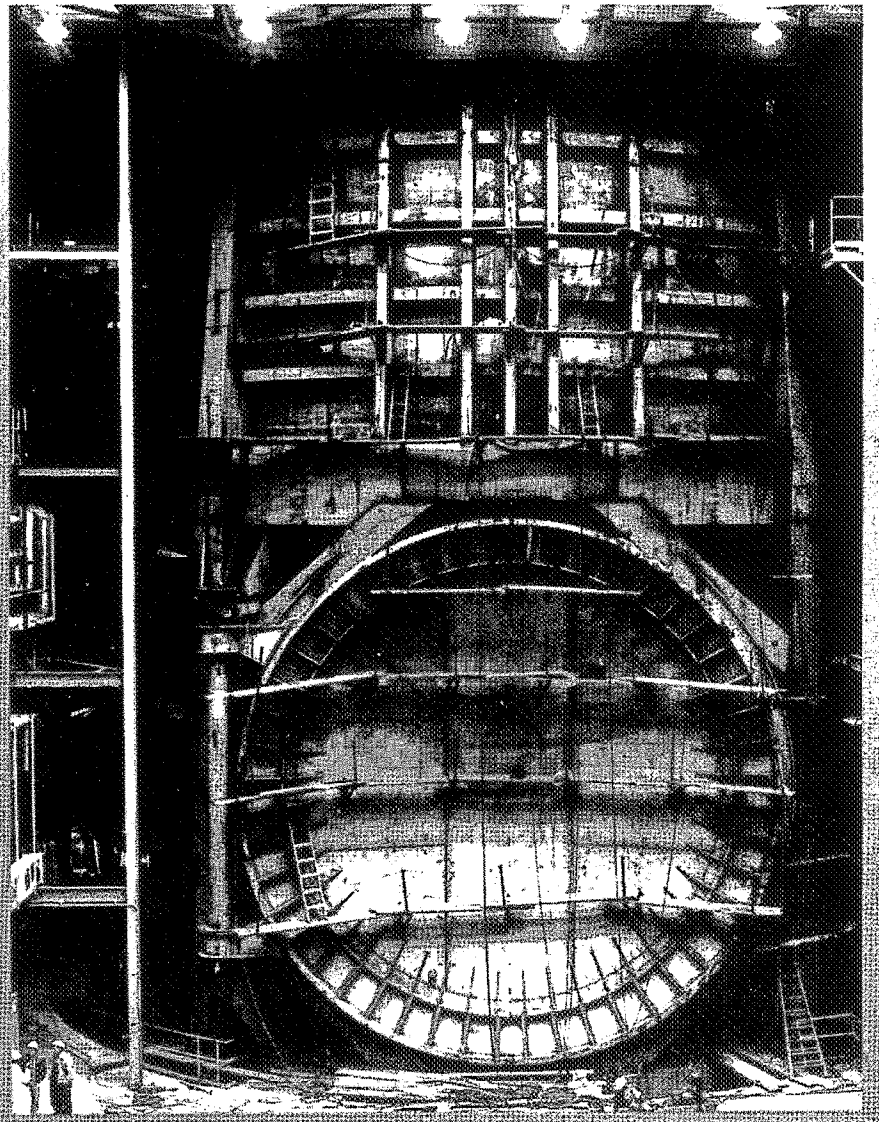


FIGURE 34

SPACE ENVIRONMENT SIMULATION LABORATORY

CHAMBER A DOOR



NASA ML65-9787
8-25-65

FIGURE 35

SPACE ENVIRONMENT SIMULATION LABORATORY
CHAMBER B

CAPABILITY & DESCRIPTION

OUTSIDE DIMENSIONS	35 FT DIA×43 FT HIGH
INSIDE CLEAR DIMENSIONS	25 FT DIA×30 FT HIGH
MAXIMUM VEHICLE SIZE	16 FT DIA×27 FT HIGH
MAXIMUM VEHICLE WEIGHT	75,000 POUNDS
PRESSURE LEVEL	1×10^{-4} TORR (70 MILES ALTITUDE)
LUNAR PLANE	STATIONARY
TEMPERATURE INTERIOR CHAMBER WALLS	-280°F
SOLAR SIMULATION SOURCE	CARBON ARC UNITS

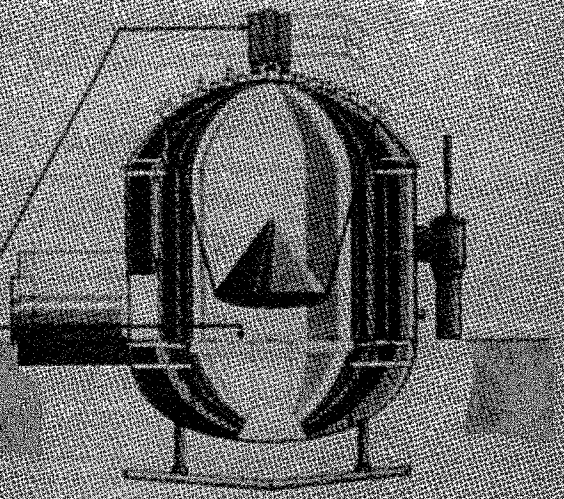
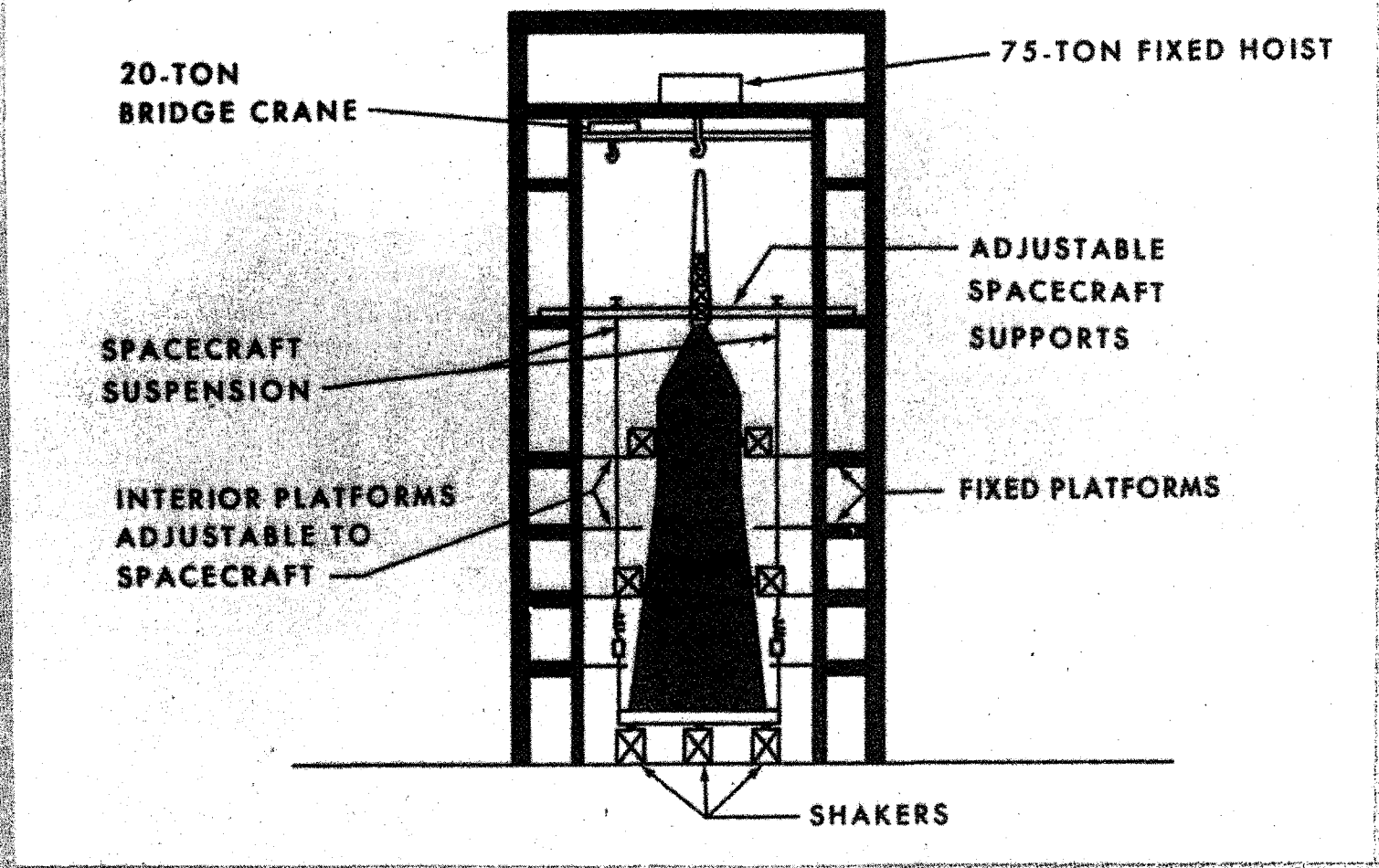


FIGURE 36

MSC VIBRATION TEST FACILITY
SPACECRAFT VIBRATION LABORATORY.



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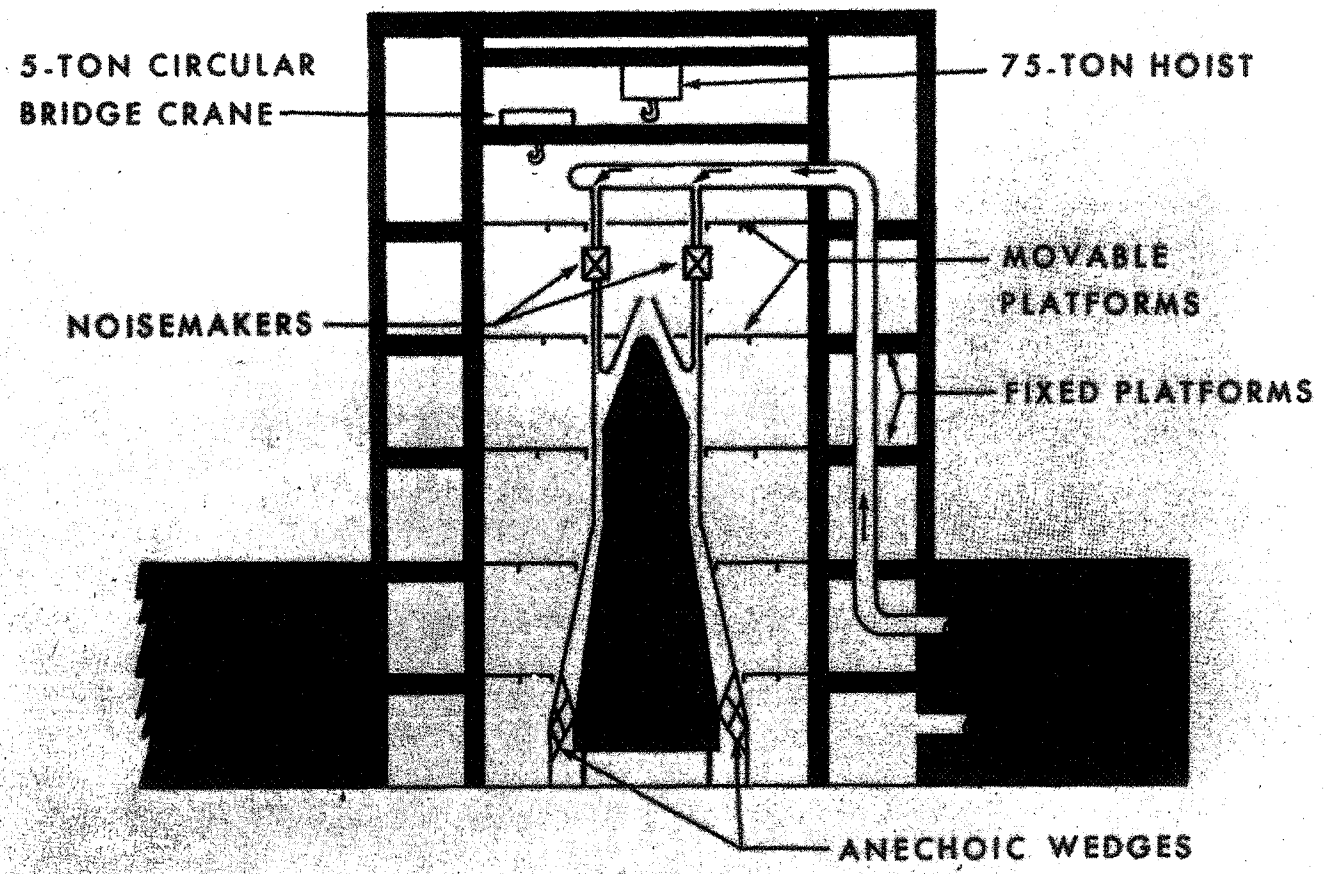
8-23-65

FIGURE 37

MSC ACOUSTIC TEST FACILITY
SPACECRAFT ACOUSTIC LABORATORY.

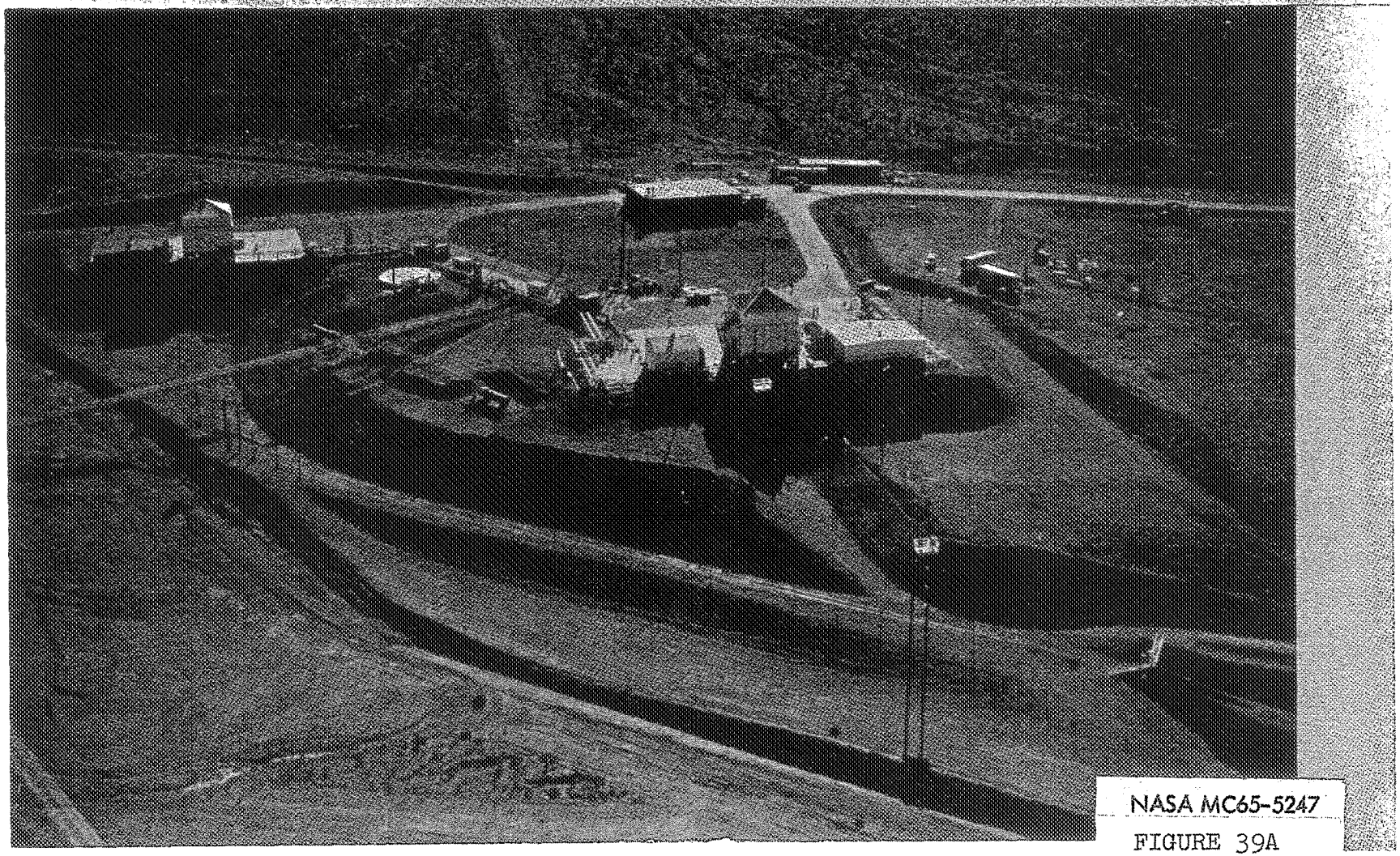


NASA ML65-9801
8-25-65

FIGURE 38

SERVICE MODULE TEST STANDS

WHITE SANDS JANUARY 1965



NASA MC65-5247

FIGURE 39A

LEM TEST STANDS
WHITE SANDS - JAN. 1965

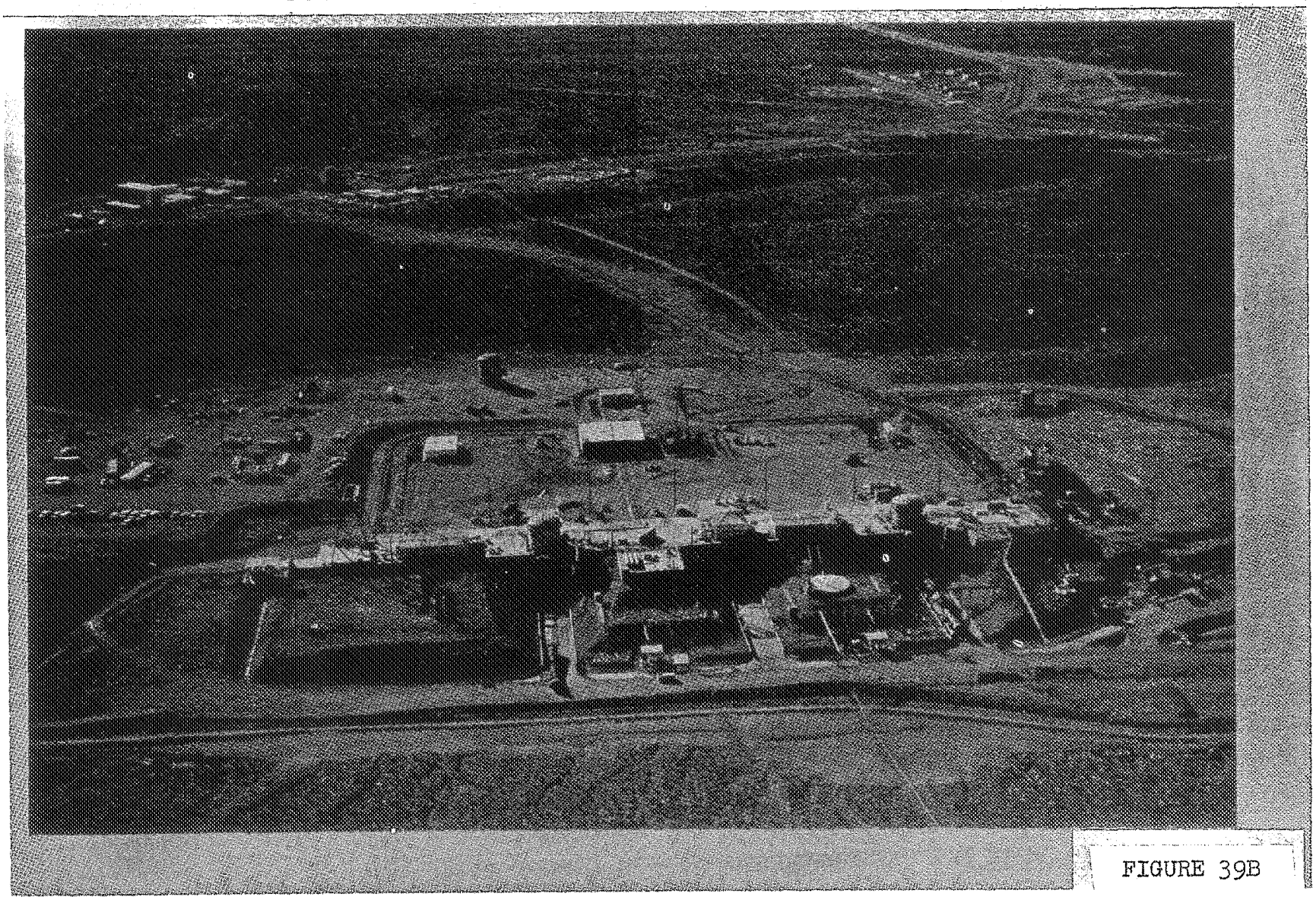


FIGURE 39B

LUNAR EXCURSION MODULE

PROPULSION SYSTEM
DEVELOPMENT FACILITY

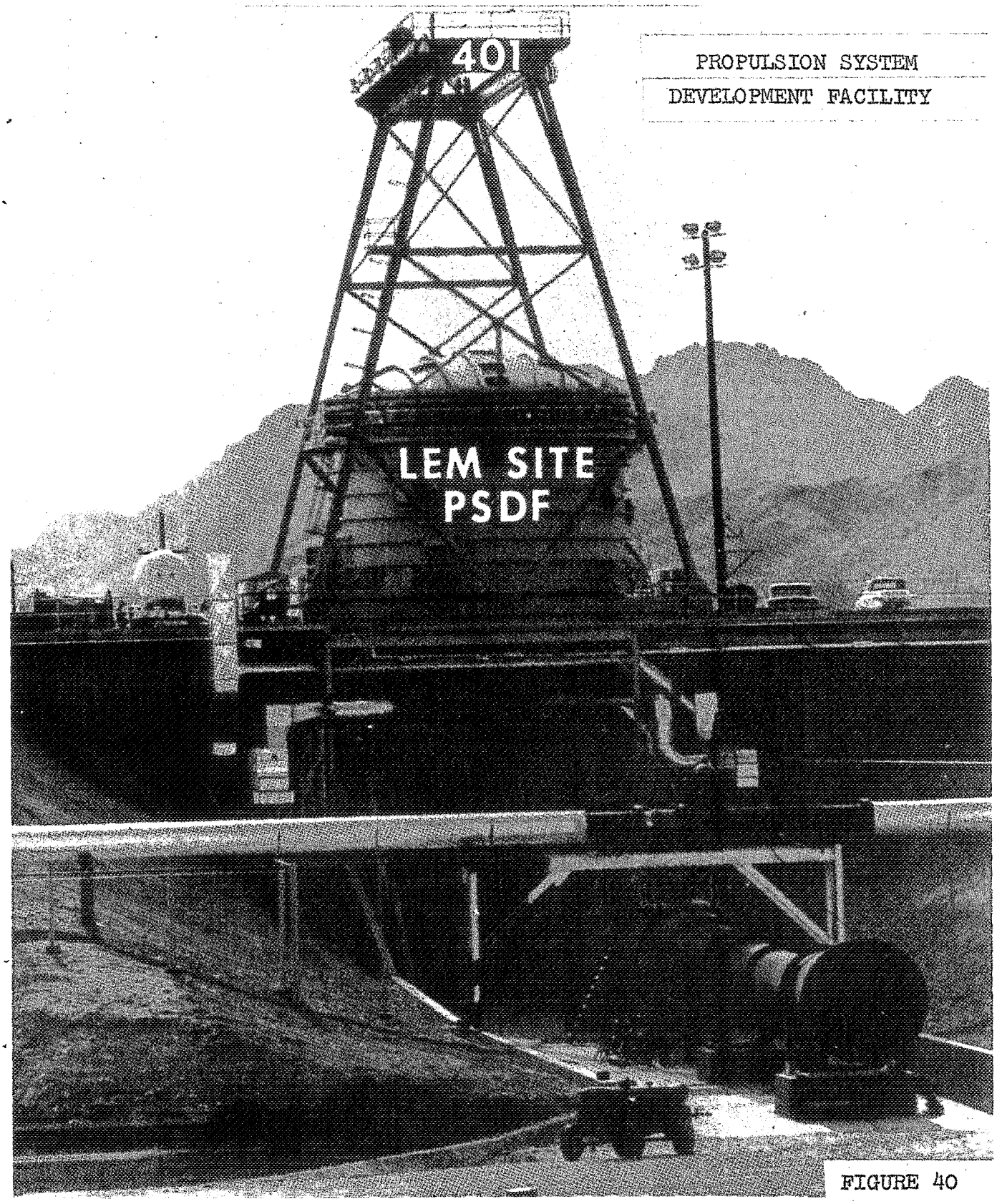


FIGURE 40

LUNAR LANDING - RESEARCH FACILITY

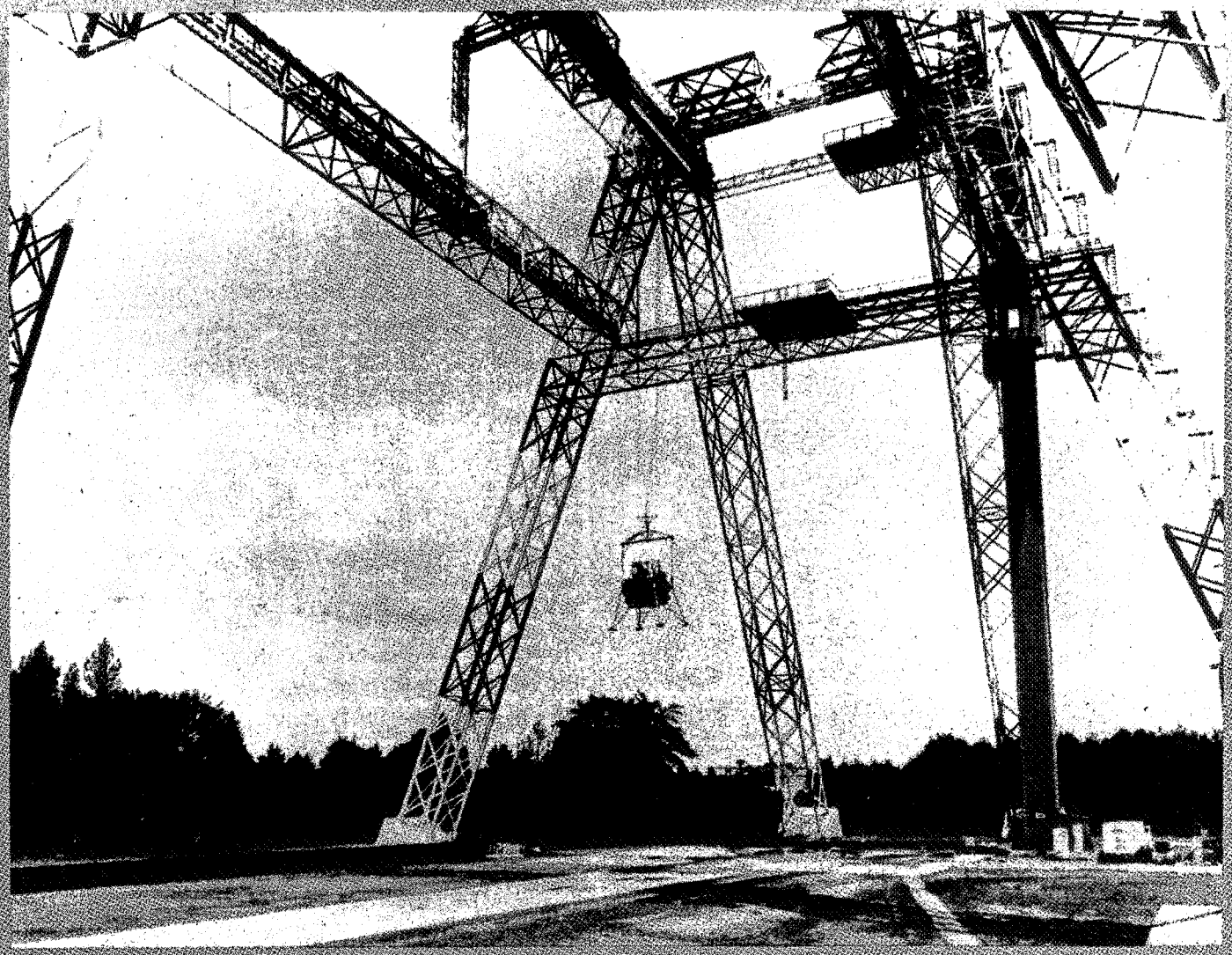


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FIGURE 41

APOLLO SPACECRAFT & SATURN IB LAUNCH VEHICLE CHECKOUT

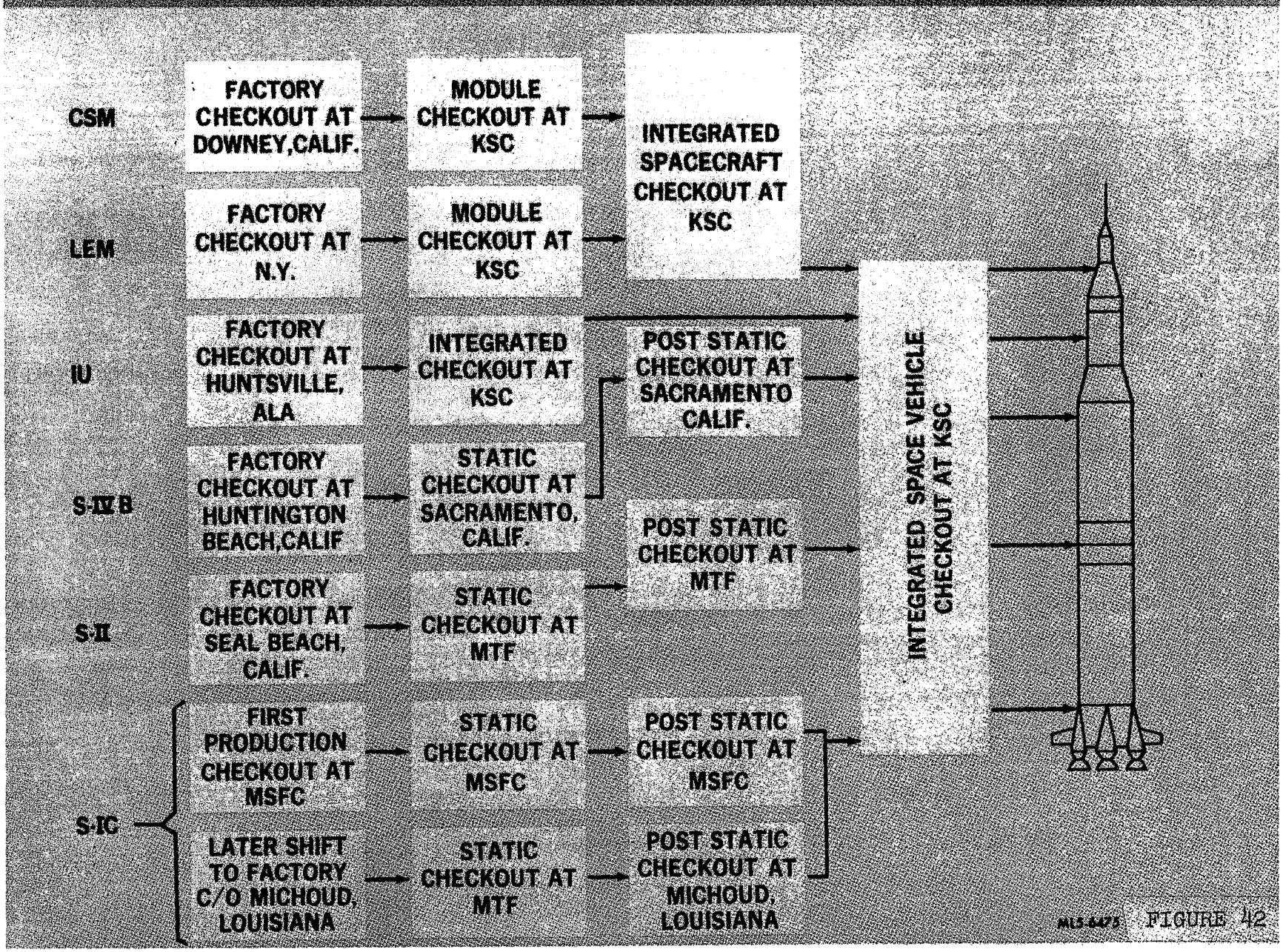

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FIGURE 42

APOLLO SPACECRAFT AUTOMATIC CHECKOUT SYSTEM

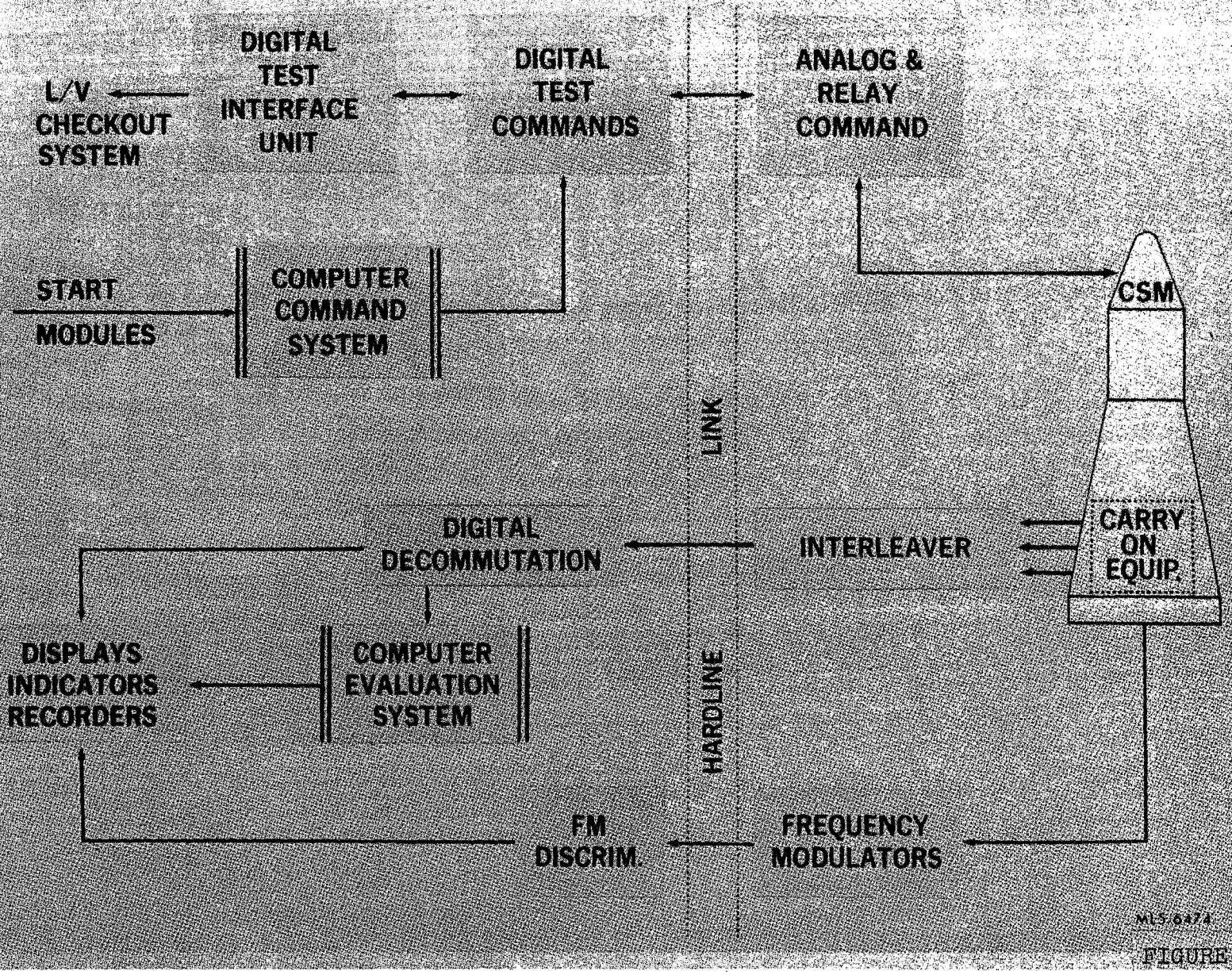

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FIGURE 43

NASA-S-64-6817

FIGURE 44

SATURN V LAUNCH VEHICLE AUTOMATIC CHECKOUT SYSTEM

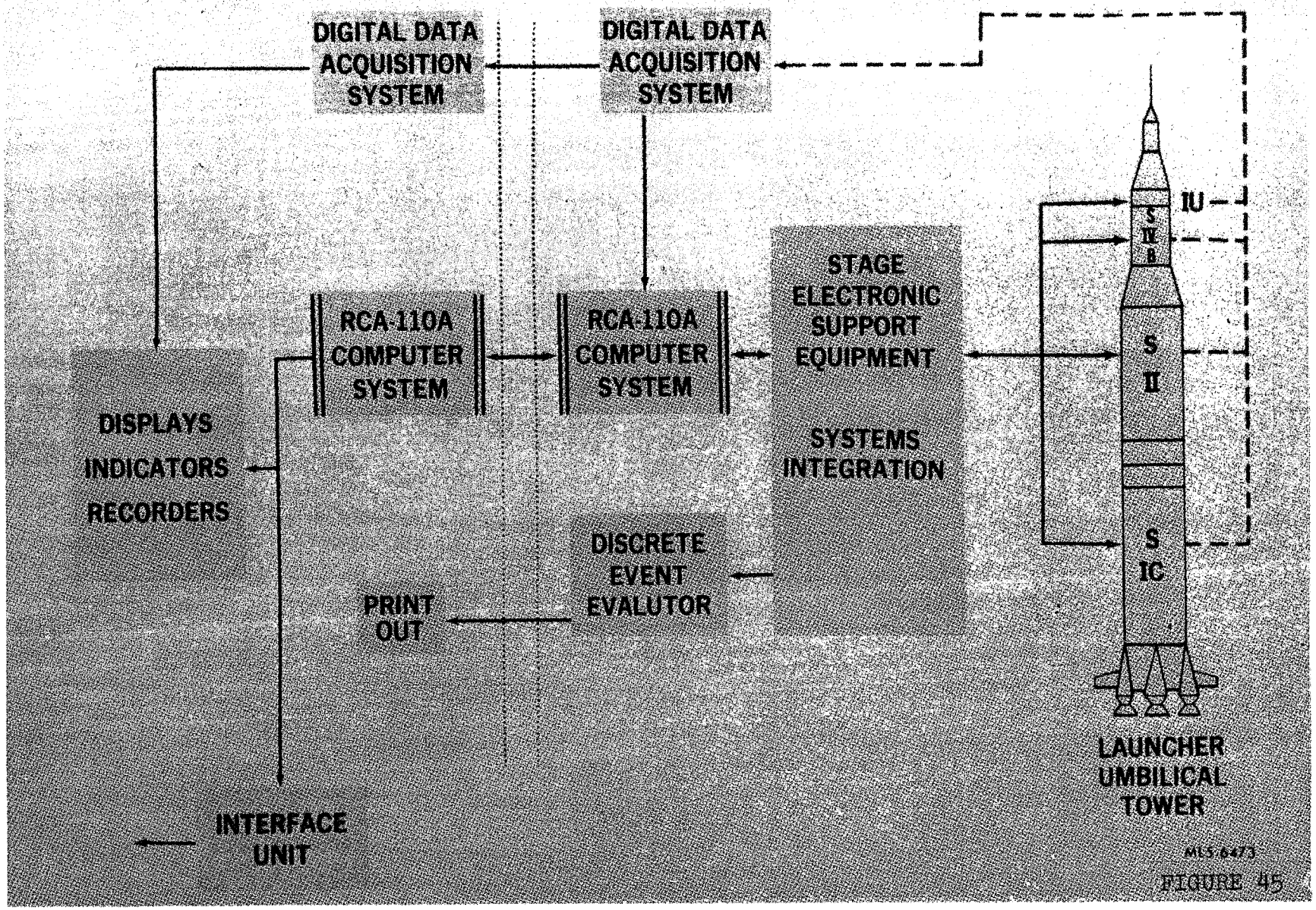

FIGURE 45

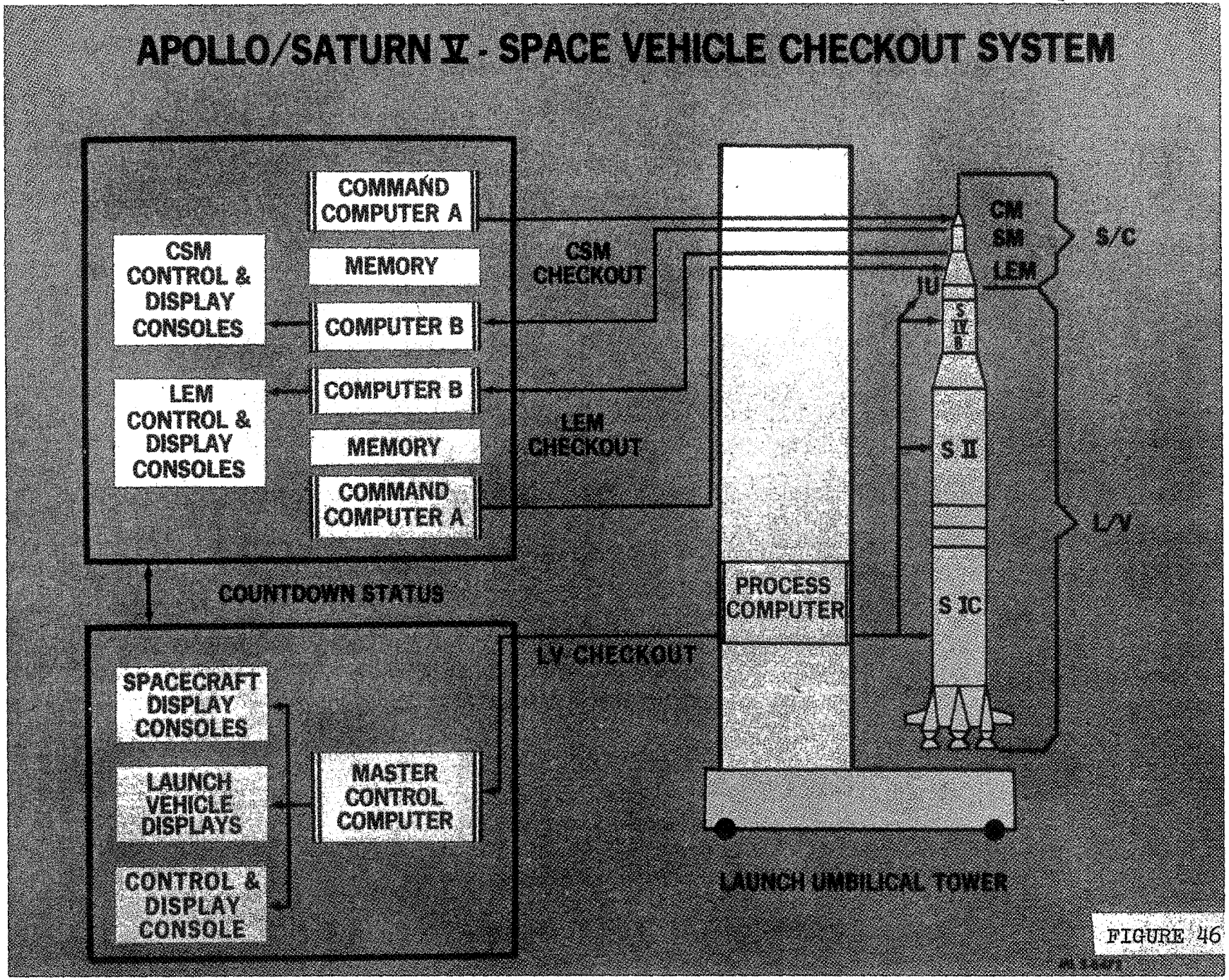

FIGURE 46

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